CRANFIELD UNIVERSITY

SCHOOL OF MECHANICAL ENGINEERING

Ph.D. THESIS

Academic Year 1996/7

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VARIABLE CYCLE PROPULSION SYSTEMS FOR A SUPERSONIC CIVIL TRANSPORT

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April 1997

This thesis is submitted in partial fulfilment of the requirements for the Degree of Doctor of Philosophy.

Summary

The different aspects of overall performance of three variable cycle engines (VCE) candidates for future supersonic civil transport are analysed in this work. These aspects concern the design and off-design points performance, the airframe engine integration and variable geometry compressor and turbine design and performance.

The three engines are compared to a traditional turbojet. The variable compressor maps were obtained with their running lines for the whole mission profile including the transition mode from medium bypass ratio to a lower bypass ratio turbofan. The specific fuel consumption (SFC) of the VCEs showed a significant improvement, especially at subsonic cruise, relative to a Turbojet engine. The extent of the variable geometry on the compressor stator angles, mixing area and the nozzle throat and exit areas is evaluated. The Fuel bill is estimated for two standard mission profiles.

The effect of installation is estimated on an isolated nacelle. A sizing calculation is carried out for the whole nacelle including the intake and the nozzle. The drag due to the friction, pre-entry, afterbody and the shock waves is calculated in order to estimated the installed performance of the three engines.

In the search of improving the VCE performance at subsonic cruise, the use of variable geometry at the low pressure turbine for the Turbofan-Turbojet engine is investigated. The effects of varying the LP turbine guide vanes stagger angle on the engine performance and component parameters are analysed. The turbine efficiency and non-dimensional mass flow changes due to the use of variable geometry are estimated.

An updated version of the Turbomatch program was corrected and tested in order to study variable cycle engines, especially to simulate the transition from one mode to another.

Acknowledgements

I would like to thank my supervisor Dr. P. Pilidis, for his help and friendship. Thanks also to Barbara Wilson for her kindness and help.

Thanks to my parents and family Maha , Rami and shadi

I would like to thank the Higher Institute of Applied Science and Technology (HIAST) for their financial support.

Finally, thanks to God who created us and gave us the hope and the intelligence to undertake such a work.

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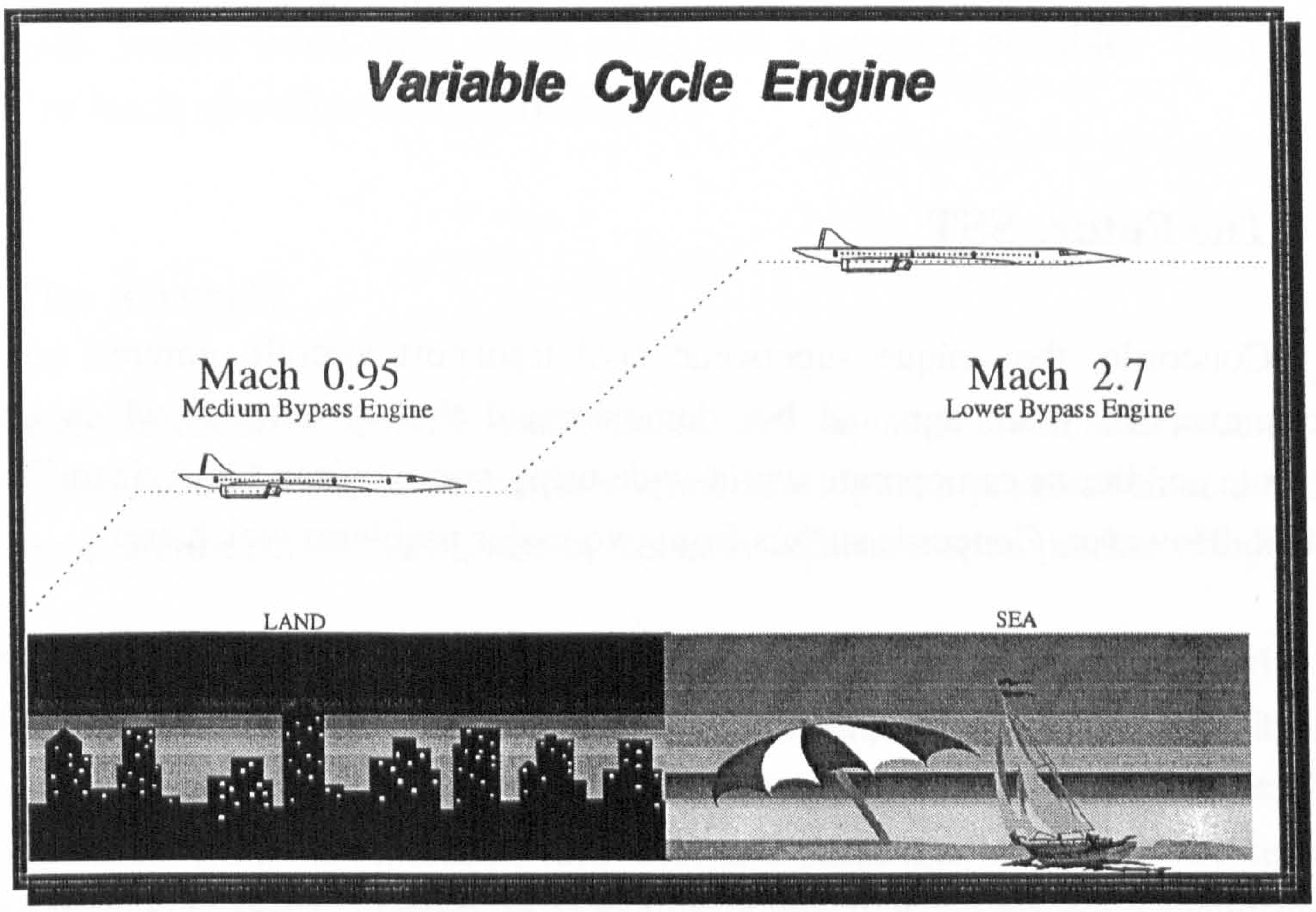
Notation

A	Area.
BPR	Bypass Ratio.
CC	Combustion Chamber.
C_f	Skin Friction Coefficient.
D	Total Drag.
DBE	Double Bypass Engine.
EPA	Environmental Protection Agency.
FAR	Federal Aviation Requirements.
FF	Form Factor.
F_N	Net Thrust.
H	Altitude.
HPC	High Pressure Compressor.
HPT	High Pressure Turbine.
IPC	Intermediate Pressure Compressor.
IF	Interference Factor.
LPC	Low Pressure Compressor.
LPT	Low Pressure Turbine.
M_o	Free Stream Mach Number.
MTF	Mid-Tandem Fan Engine.
N	Rotational Speed.
NDMF	Non Dimensional Mass Flow. ($W.\sqrt{T}/P$) (kg. $\sqrt{K}$/Pascal)
N/Engine	Newton per Engine.
NGV	Nozzle Guide Vanes
OPR	Overall Pressure ratio.
P	Total Pressure.
q	Dynamic Head Pressure.
R_{NL}	Reynolds Number Based On Component Length.
S	Nacell Wetted Area.
SFC	Specific Fuel Consumption.
SST	SuperSonic Transport.
T	Total Temperature.
t	Static Temperature.
TET	Turbine Entry Temperature.
TFTJ	Turbofan-Turbojet Engine.
U	Blade Rotational Speed.
V_o	Air Free Stream Velocity.
W	Mass Flow

β	Shock Angle.
ϕ	Component Drag, Flow Coefficient.
η	Efficiency
θ	Ramp Angle.
ρ	Air Density.
ψ	Blade Loading Coefficient.

CHAPTER 1

Introduction



The major challenges with the design of propulsion systems for high speed civil transport are complying with FAR and EPA environmental standards while powering an economically acceptable aircraft. These issues create a dilemma in engine design because low exhaust jet velocities are required to meet takeoff noise regulations while high exhaust jet velocities are required for economical supersonic cruise operation. Moreover, good performance is needed for long subsonic cruise. Previous studies have shown that to meet FAR stage 3 noise regulations, engines incorporating mechanical/thermal noise suppression concepts must be oversized by 50% to 80% relative to the size that will provide the maximum aircraft economics.

1.1- The Future SST
1.2- Aim Of The Work
1.3- Bibliographic Review

CHAPTER 1

Introduction

1.1 The Future SST

Concorde, the unique supersonic civil transport aircraft, entered commercial services over 15 years ago and has demonstrated that an aircraft which cruises at supersonic conditions can operate world-wide using normal airport and air traffic control facilities. However, Concorde suffers from two major problems which are:

1- It is not an economically successful aircraft. The fuel consumption of its engine Olympus 593 at subsonic cruise is high in comparison with the modern subsonic aircraft engines.

2- It does not meet the actual environment requirements. Especially at take-off where the noise level is unacceptable.

A new supersonic transport will need to be an economically successful program. It has to compete with tomorrow's advanced subsonic aeroplanes. Environment issues will be a key factor in SST acceptance and most of the answers are in the engine manufacturers hands.

1.1.1 The Market

Recent studies in the U.S., Europe, and Japan indicate significant market potential for a new high-capacity supersonic transport linking the world's major cities (Ref. 1). This SST could be economically viable and environmentally acceptable, provided investments are made to acquire the required technologies. Development costs during the next decade are estimated at $15 billion. The SST would fly over the

Atlantic, the Pacific, and desert areas, covering about 80% of the most attractive routes, where supersonic flight (with its propagated boom) is allowed. In the year 2010, long-haul traffic is estimated at 300 million passengers/year, at least a third of whom could be persuaded to fly supersonic. Fares for supersonic flights must not be more than 20% above those for subsonic travel.

The size of the market, estimated as being between 500 and 1000 aircrafts, shows that there will be room for only a single program; only extensive and balanced international cooperation combining the skills and the facilities of the main manufacturers and research centers world-wide would make such a program possible. The aircraft is expected to reach operational status by 2005-10.

1.1.2 The Aircraft

Right selection of aircraft specification such as number of seats, cruise Mach number, range, is the key to a successful aircraft development program since it has large impact upon aircraft performance, hence, has large influence upon market viability of the aircraft.

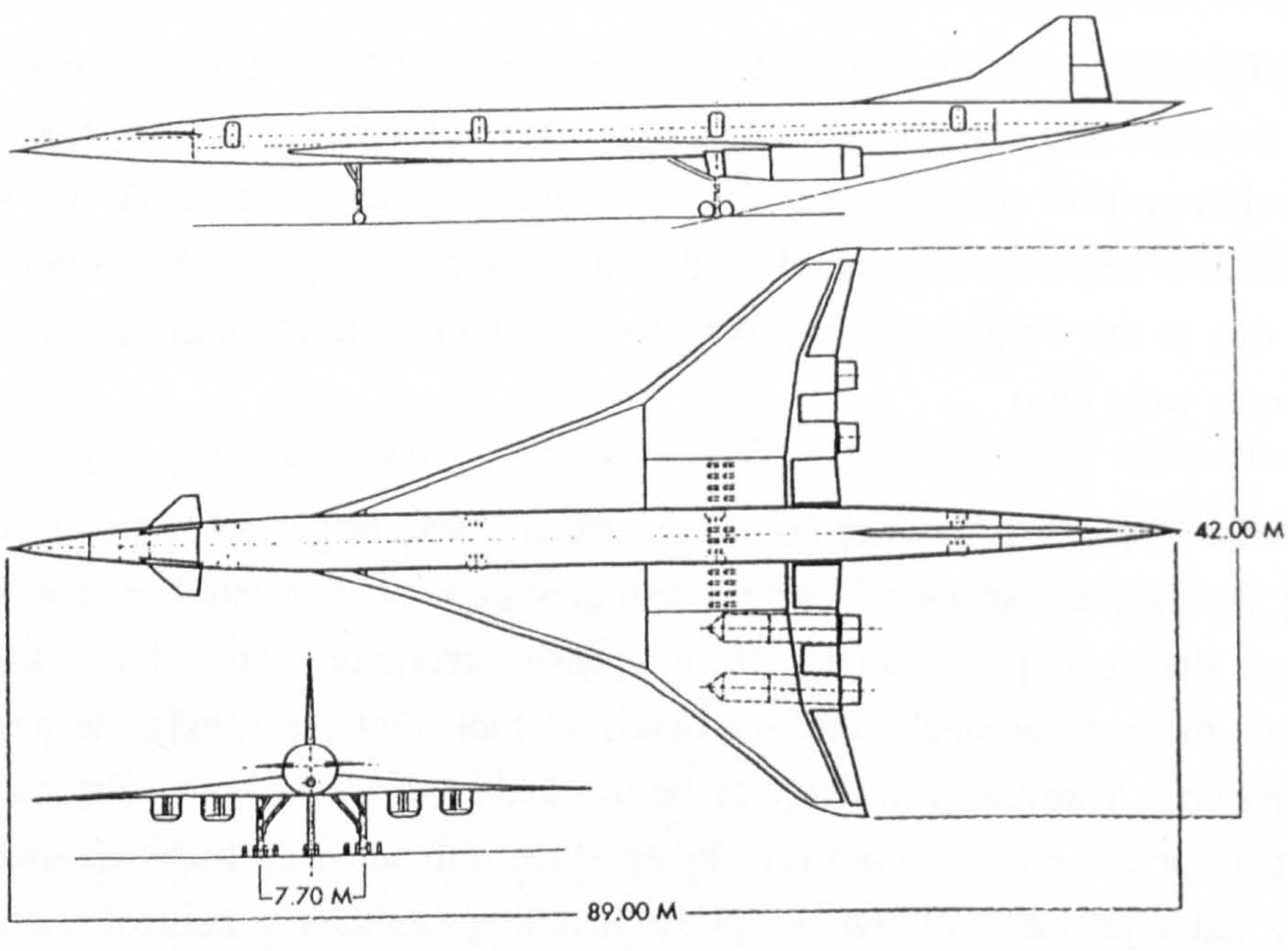
Many different studies,(Ref. 2, 3, 4), agreed that the future supersonic aircraft will have between 250 and 300 passengers in a three class seating arrangement. This is the best solution from standpoint of fare level and passenger share. A 400-seater had been discarded because, compared with 300-seaters, its passenger share was not increased due to the weight and drag penalty of its twin aisle cabin arrangement and the required units were less.

Cruise Mach number may be the most controversial parameter to be selected in designing supersonic transport. Previous studies have concentrated on the Mach 2~3 speed range due to requirement for an inexpensive thermally stable fuel. Reference 2 showed that the conventional Kerosene-based jet fuels that can satisfy the requirements of up to Mach 2.8 aircraft are likely to be available for use in both subsonic and SST aircraft at no price penalty. Fuel for higher Mach number need to be thermally stable. Thermally stable jet fuel will involve special handling facilities to control contamination, this may necessitate additional airports infrastructure, affecting the cost of the fuel and the market viability monotonously decreases as cruise Mach number is increased beyond that range. Lower cruise Mach number airplanes can offer lower fare but it does not improve the passenger share because the penalty of higher fare for higher speed airplane is compensated by shorter trip time and larger stimulation effect.

The range requirements of SST would be governed by the need to satisfy key pacific markets. The Los Angeles-Sydney route represents the longest range, 6500 Nautical miles, and is over water and thus could be carried out supersonically. A future SST airplane will be required to operate some significant portion of its mission at subsonic cruise speeds, over areas where the sonic boom that a SST airplane would generate would not be tolerated.

Since long distances will be completed in a fraction of the time of subsonic aircraft, weather predictions will be more reliable, thus minimising flight delays and rerouting which will have a favourable effect on fuel reserve planning. At cruise altitude of +18000 meters the influence of winds on range is less than 10% that of a conventional subsonic aircraft, with cruise altitude around 9000~13000 meters.

Figure 1.1 shows an example of future SST airplane. The take-off total weight is estimated at 350,000 kg. It will have aerodynamics optimised for both supersonic and subsonic cruise (Mach 0.95). The drag at low speeds will be reduced by using advanced lift augmentation devices. Furthermore, the airframe and the engine nacelle structural weight will be reduced by using aluminium and titanium alloys and by the large-scale introduction of composites.



The joint Aerospatiale/British Aerospace/ Deutsche Aerospace configuration features a 5,500-n.mi. range, a maximum takeoff weight of 320 tonnes, and will carry 250 passengers.

Figure 1.1 Future SST Aircraft.

Now, available calculation methods and use of slat and elevon deflections will enable better aerodynamic efficiency at all flight speeds while optimizing the wing for supersonic cruise. Emphasis will be on the best possible aerodynamic efficiency in cruise in high subsonic range, to make extensive flight without sonic boom above inhabited territories economically viable.

The friction drag represents more than 40% of the total aircraft drag in supersonic cruise. Its reduction is a long-term objective; solutions are now within the reach of research centers (Ref. 1). It is anticipated that a hybrid laminar boundary layer control on the wing could reduce aircraft friction drag by over 30%, and lead to an increase in operating range by more than 10%.

1.1.3 The Engine

The new SSTs would have to meet the same noise and pollution regulations as subsonic aircraft; they could not be given special status. A new SST would certainly have to conform to stage 3 noise regulations. Low emissions would be a vital aspect of a new SST engine. A massive reduction in oxides of nitrogen is expected.

The conflicting needs of an SST powerplant include a low bypass ratio for supersonic cruise, but with the ability to meet take-off noise requirement without an ejector device and without requiring a large fan diameter.

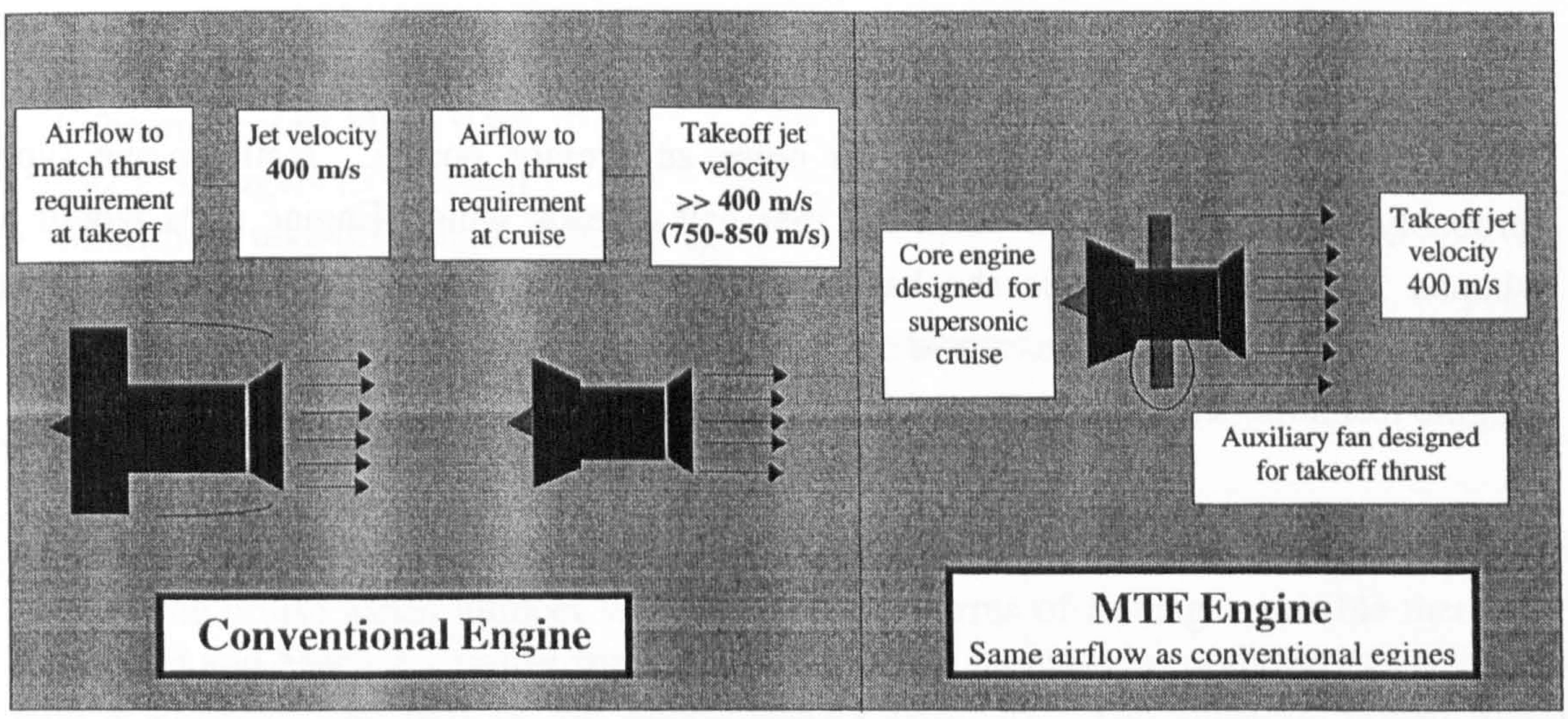


Figure 1.2 Comparison of MTF And Conventional Engines.

Rolls-Royce and Snecma joined by MTU in Germany, and Fiat in Italy have been working on an SST powerplant since early 1990. In their view, the appropriate configuration would be a mid-tandem fan (MTF) variable-cycle engine, because it reduces the compromises of an SST powerplant, providing a turbofan for take-off and subsonic flight and a turbojet for supersonic cruise (Figure 1.2).

This design is based on the insertion of the fan down stream of the low-pressure compressor, where the primary airflow is already compressed in a duct of moderate diameter. The result is an engine with an acceptable frontal surface area of around 2 m in diameter. This secondary fan therefore plays a dual role: it supplies the bypass flow in its external section and it contributes to compressing the air in the primary circuit (between the low-pressure and high-pressure compressors) in its inner section. This concept results from a trade-off between current subsonic turbofan and the minimum-cross-section nacelle needed to reduce drag at supersonic cruise.

On the other hand, as part of the SST propulsion and airframe system studies, primary and alternative engine concepts have already been selected from eight possible engine types by the combined NASA, GE, Pratt&Whitney, McDonnell Douglas and Boeing team (Ref. 7).

A small diameter, relatively low bypass mixed-flow turbofan is the first choice. It, however, depends on the successful mixing of the engine and the fan air and the development of a variable geometry engine noise suppressor nozzle.

The alternate engine concept, a higher bypass, fan-on-blade, or "flade" approach, has a higher bypass ratio. Its wider diameter would negatively affect aircraft aerodynamics and overall efficiency. A flade engine, however, would require a significantly lower weight or even no noise suppressor nozzle. Both engine types probably would be mounted under the rear half of each wing. Engine pods would be shaped for favourable sonic shocks wave interference. Locally tailored lower wing surfaces and variable inlets would minimise and control inlet flow distortion. Both engine technologies should be capable of meeting the current Stage 3 noise standards.

1.2 Aim Of The Work

The main aim of this work is to study the different aspects of overall performance of three variable cycle engines selected for powering the future SST aircraft. These aspects concern the design and off-design point performance, the airframe engine integration, compressor and turbine design and variable geometry and the development of a methodology to match variable cycle engine. These engines are the following:

Turbofan-Turbojet
Mid-Tandem Fan
Double Bypass Engine

Market studies have highlighted that a successful SST should have:

- Supersonic cruise Mach 2~3
- 200~300 passengers
- Range of 6500 Nm

A literature search was conducted to identify aircraft configurations that satisfied these requirements. Many studies have been carried out over the years but none corresponded with the requirements outlined above, there were variations on range, cruise Mach number and number of passengers. A NASA report (Ref. 5) contained enough information on the airframe to allow the original configuration to be modified to meet our outlined requirements. The original concept was a 292 passenger, Mach 2.7 design utilising four dry turbojets as its propulsion system. The modified aircraft configuration had the following design requirements.

- Supersonic cruise Mach 2.7
- Subsonic cruise Mach 0.95
- 292 Passengers (3 Classes)
- Maximum take off weight 350,000 kg
- 6500 Nm range
- Balanced field length < 3170 meters. Sea level standard day +15 K.
- 1.33 Thrust margin at take-off
- FAR-121:648 (modified) fuel reserves
- FAR 36 Stage III noise regulation

The cruise Mach number was acceptable in terms of finding a suitable thermally stable fuel and Mach 2.7 cruise speed showed a 150% increase in productivity compared with a subsonic transport of the same capacity (Ref. 8). The subsonic cruise Mach number was 0.95, with the limitation being the onset of supersonic wave drag. The subsonic cruise Mach number was higher then a conventional subsonic aircraft, which is usually mach 0.85, due to the slender delta wing airframe design. The 292 passengers were carried in three classes, first business and coach class, with the number in each class dependent upon operating economics. In addition to the 292 passengers and their baggage there would also be 14.2 m^3 of cargo with a maximum weight of 3,994 kg this would give a maximum total payload of 350,000 kg

The thrust requirements, for a four engine aircraft, to maintain a rate of climb of 1,000 m/min (3,000 feet/min), are:

Mach Number	Altitude(m)	Accelerating Climb
0	0	181,500 N/Engine
0.5	460	191,500. N/Engine
0.6	2290	146,750 N/Engine
0.95	9150	119,000 N/Engine
1.05	9750	126,500 N/Engine
1.6	12250	133,000 N/Engine
2.7	18750	100,000 N/Engine

The cycle of the engine has some physical constraints that will limit its performance. The main limitation to the overall pressure ratio was the maximum allowable compressor exit temperature. Due to the large amount of ram compression, producing a large temperature rise, experienced at high Mach number flight conditions the compressor would be operating at high compressor exit temperatures for long periods. Due to this operation at high temperature for long periods the material creep properties become increasingly important. The temperature limit at which creep properties of Titanium alloys would be acceptable, for the year 2000+, was estimated to be around 950 K.

The VCE will operate at high turbine entry temperatures for long periods of time, during supersonic cruise, and therefore good high temperature creep properties are required to allow the design of a turbine with an acceptable commercial life. A limit of 1800 K was set on the turbine entry temperature due to material creep properties. These long periods of operation at high turbine entry temperature require good combuster exit temperature profile and commercially acceptable combuster life. The main combuster pressure drop was assumed to be 5% with a combustion efficiency of 99.5%.

Due to the high turbine entry temperatures 15% of high pressure compressor bleed air was required for turbine blade cooling. The maximum duct burner exit temperature was limited to the value of 1700 K to provide good commercial life.

Compressor and turbine isentropic efficiencies were assumed to be 88% and 90% respectively. These levels of turbomachinary efficiencies were not considered to be unduly challenging. In order to obtain a meaningful comparison between the engines, the following parameters were used:

Maximum Turbine Entry Temperature (TET)	1800 K
Maximum HP compressor outlet temperature	950 K
Compressor Isentropic Efficiency	88%
Turbine Isentropic Efficiency	90 %
Combustion Efficiency	99.5 %
Reheat Efficiency	90 %
Reheat Ducts Pressure Loss	3 %
Turbine Cooling Mass Flow	15 %

The performance calculations were carried out by using an updated version of the TURBOMATCH program (Ref. 9) which is widely used at Cranfield University. This was supported by other software developed at the University, to design the compressors and turbines respectively (Ref. 10). The compressor off-design performance and the variable geometry compressor maps were obtained by using a compressor off-design analysis program (Ref. 10).

The effect of engine integration to the airframe was carried out in Chapter 6 and the engines installed performance was obtained. Moreover, the effect of using a variable geometry in the low-pressure turbine for the Turbofan-Turbojet engine was investigated in Chapter 5. Chapter 3 was reserved for the design point analysis of the three VCEs. The timetable for the whole work during the three years was as follows:

Bibliographic Review	6 months
Turbomatch Modification	6 months
The Three Engine Simulation and Components Design	12 months
Engine Airframe Integration	5 months
Variable Geometry low-pressure Turbine	4 months
Holidays	3 months

1.3 Bibliographic Review

There are very few works published, in the public domain, on variable cycle engines. The majority of the work has dealt with different design concepts and potential applications. There is very little previous work assessing the component performance and feasibility of such engines. The review carried out herein intends to show the latest tendencies and analysis accomplished so far of some VCEs concepts both at Cranfield University and world-wide.

1.3.1 *Cranfield University*

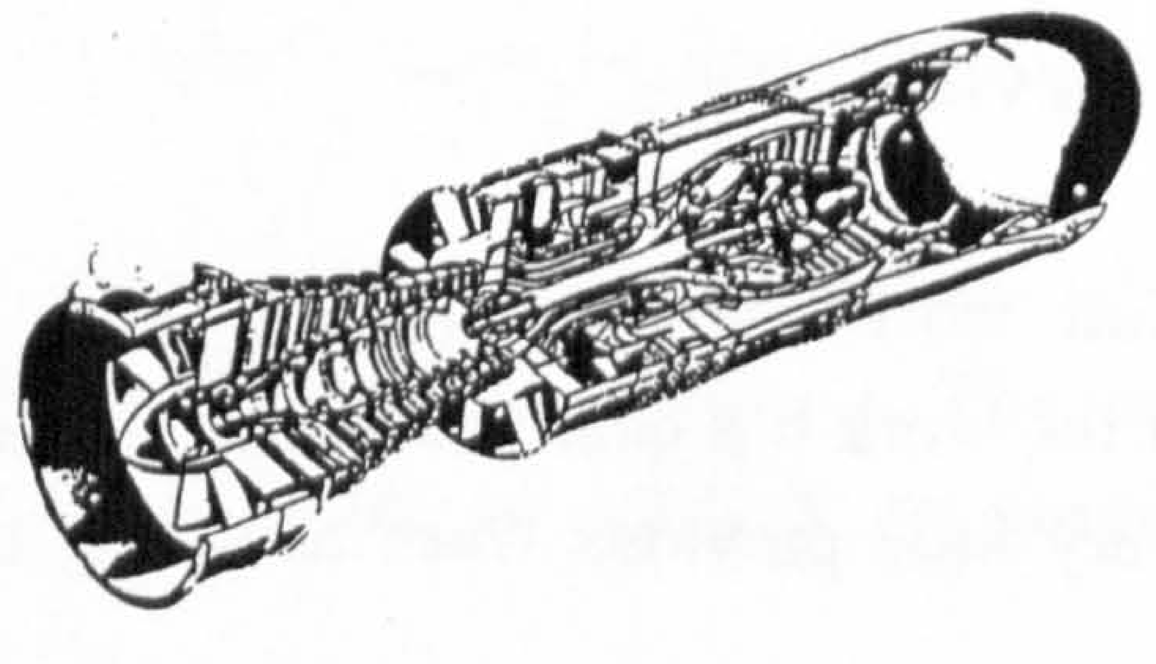
M.A.R. Nascimento, 1992: In this work, a variable jet engine is investigated for supersonic (Mach 1.6) ASTOVL military aircraft. This engine is a Selective Bleed Variable Cycle, twin shaft turbofan. The performance of the engine and its components is analysed using a novel matching procedure. Off-design engine performance characteristics are explained, compressor running lines are shown, and variable geometry requirements are described. The benefits predicted by this preliminary investigation indicate that further research on this engine should be carried out (Ref. 10).

J.E.A. Roy-Aikins, 1988: A general investigation of the advantages of using variable geometry in the key gas turbine engine components was carried out in this work. A computer program was developed to simulate the steady state performance of arbitrary gas turbines with or without variable geometry in the gas path components. The program was used to study some cycles incorporating variable geometry and it was concluded that variable geometry can significantly improve the off-design performance of gas turbines (Ref. 11).

1.3.2 *Europe*

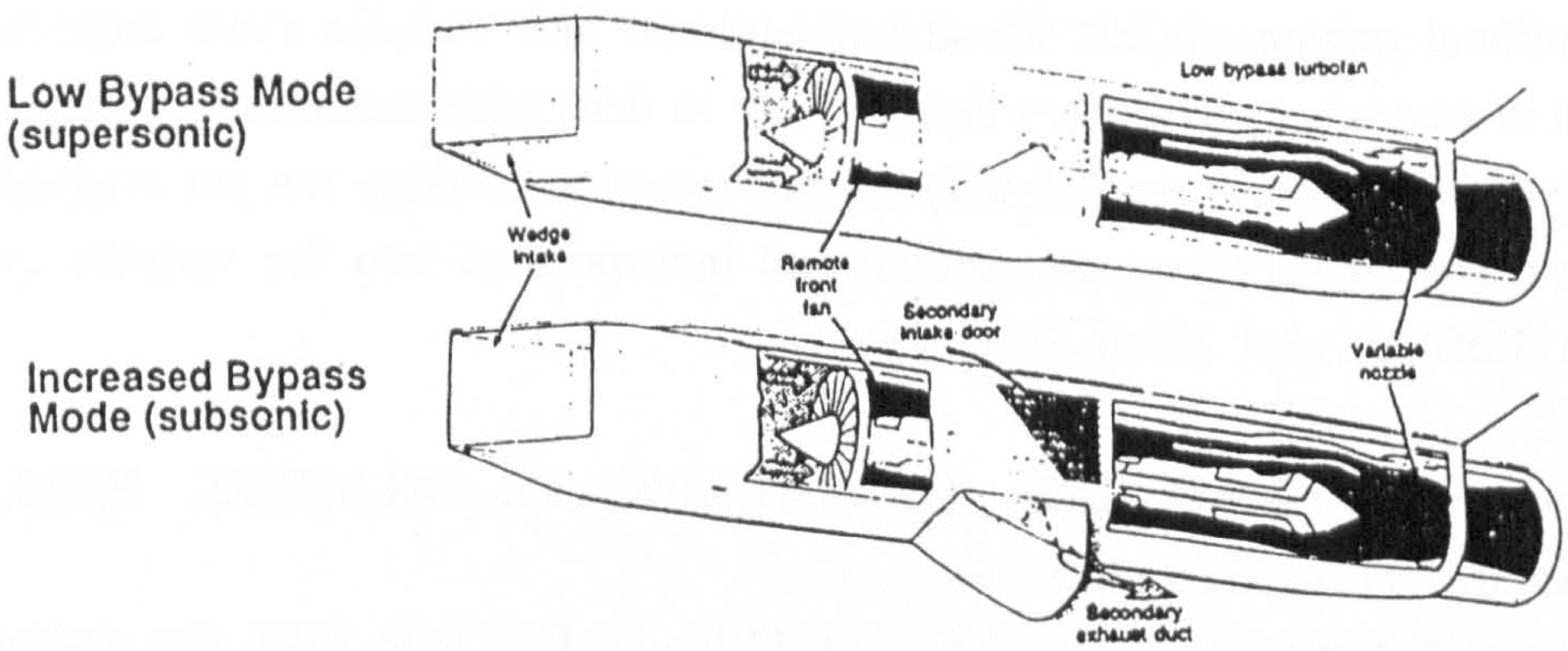
Snecma, France. MCV99 Engine: It is a two spool engine. The main spool is a classical turbojet. A secondary spool, consisting of an additional fan driven by a turbine is placed in the middle of the main spool, where the primary gas stream is at its smallest, causing a minimum increase of the overall diameter of the engine. At take-off and climb the secondary fan is fed by auxiliary intakes, increasing the total air flow through the engine. The necessary power to drive this fan is obtained by bleeding air from the main compressor to the secondary turbine. In supersonic cruise, the secondary intakes and the HP air bleed are closed, and the engine operates as a turbojet (Ref. 12).

SNECMA - MCV 99

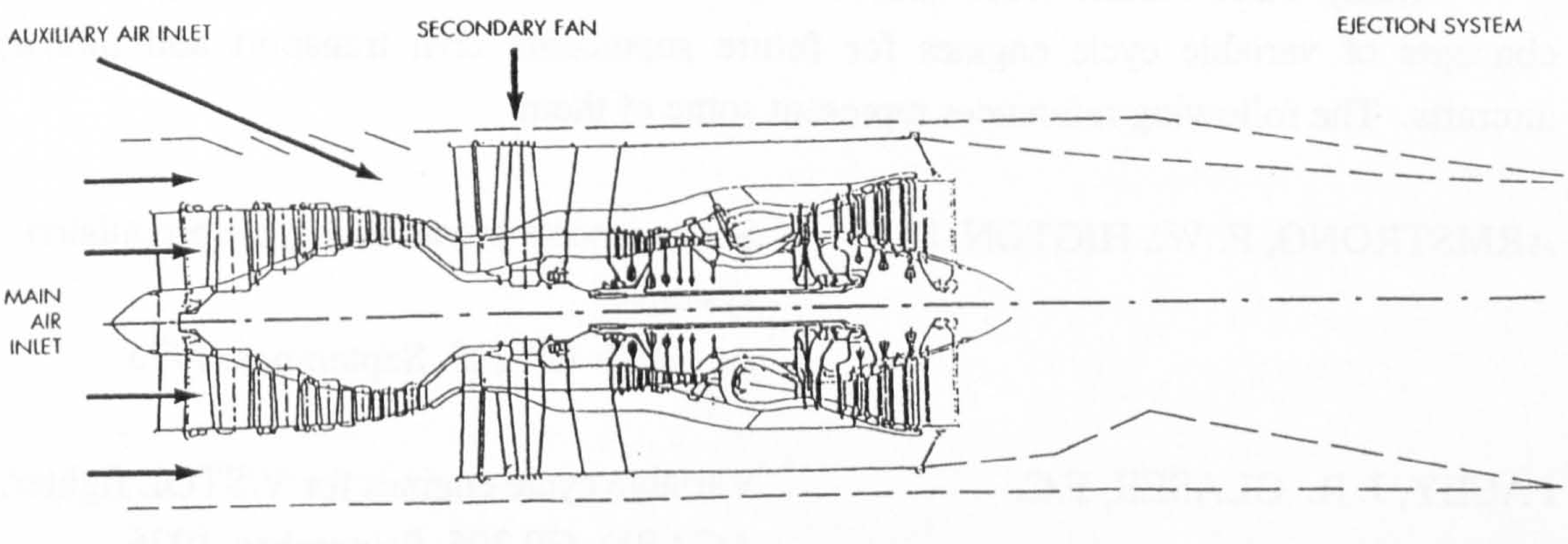


Rolls-Royce, England. Tandem Fan Engine: This engine consists of a classical twin spool turbofan in front of which there is an additional placed front fan. This later is also driven by the LP turbine of the basic turbofan. In supersonic cruise, the air flows through both secondary and main fan. The engine acts then like a classical turbofan with moderate bypass ratio. During take-off and climb, a "mode change valve" mechanism ejects the air coming from the main intake and the front fan, via an auxiliary nozzle and an auxiliary intake feeds the rest of the engine, thus increasing the total airflow (Ref. 12).

ROLLS ROYCE TANDEM FAN



Rolls-Royce England, Snecma France, MTU Germany, FIAT Italy. MTF Engine: The mid-tandem fan engine (MTF) benefits from the Snecma MCV-99 and Rolls-Royce tandem fan concepts. It has two flow paths having a bypass ratio greater than 2 for take-off and climb. The bypass flow enters the engine via both the frontal air inlet and the lateral air inlets. This double flow path, which provides low specific fuel consumption and reduce noise levels due to the moderate ejection speed (about 400 m/sec at the nozzle outlet), allows noise regulations to be met. With a reduced bypass ratio of 0.7, favourable for supersonic cruise. Here, lateral inlets are closed and the variable-pitch guide vanes of the midfan reduce frontal airflow into the bypass duct; jet speed is then about 650 m/sec at the nozzle outlet (Ref. 12, 13).



The mid-tandem fan variable-cycle engine, shown in take-off mode, is equipped with a mid-bypass fan coupled to the secondary body.

1.3.3 *USA*

In the USA, NASA, Pratt & Whitney and General Electric have narrowed the field of five candidates, selecting in mid-October 1993 a preferred and a backup or alternate concept. While the concepts chosen are not being discussed publicly for now, they are expected to include a low bypass turbojet and turbofan.

R. D. ALLAN, General Electric, NASA CR-134913, June 1992:
This study was designed to select the best conventional mixed-flow augmented turbofan engine by a parametric assesment, and add selected variable cycle features to this conventional engine cycle. These conventional and variable cycle engines were the subject of an engine preliminary design study to determine mechanical feasibility, confirm weight and dimensions, and identify the necessary technology not yet available. Critical engine components also were studied and incorporated into the variable cycle engine design (Ref. 14).

G. A. CHAMPAGNE, G. E. ALLEN, M. PALMIERI, R. M. ADLER, NASA:
In this paper, presented at the ASME in Orlando, FL June 1991, the authors examine the use of devices that increase the engine airflow by 120% during take-off in order to reduce the exhaust velocity to acceptable levels, while sizing the main propulsion system to achieve the best possible performing aircraft. The concept evaluated incorporates an inlet flow valve to increase the engine inlet flow by 74%, and an ejector nozzle to increase the engine exhaust flow by 46%. The summary of that work was that incorporating of an inlet flow valve and ejector in a SST engine presents a viable solution for achieving acceptable exhaust jet noise levels during take-off operation. Use of mechanical noise suppresser is ineffective at the low velocities required to achieve FAR Stage 3 regulations (Ref. 15).

Many other studies were carried out in the mid seventies concerning different concepts of variable cycle engines for future supersonic civil transport and military aircrafts. The following references represent some of them:

ARMSTRONG, F. W., HIGTON, D. R.	Some aspects of variable cycle propulsion systems. AGARD CP 205, September 1976
FACEY, J. R. GLASER, F.C.	Variable cycle engines for V/STOL fighter. AGARD CP 205, September 1976

WILLS, E	Variable cycle engines for supersonic cruise aircraft. AGARD CP 205, September 1976
BOXER, E, MORRIS, S.J., FOSS, W.E	Assessment of variable cycle engines for supersonic transport. AGARD CP 205, September 1976
PAYZER, R.J.	Variable cycle engines applications and constraints. AGARD CP 205, September 1976
ALLAN, R.D.	General Electric company variable cycle engine technology demonstrator programs. AIAA No 78-1047, July 1978
BROWN, R.H.	Integration of a variable cycle engine concept in a supersonic cruise aircraft. AIAA No 78-1049, July 1978

CHAPTER 2

Variable Cycle Engine Philosophy

A turbojet engine is the best candidate for supersonic cruise. A turbofan with high bypass ratio is the most efficient at subsonic cruise. Is it possible to manufacture one engine using variable geometry features which is able to give the same performance as the above engines? In other words, is it possible to match a turbojet cycle with a high bypass ratio turbofan cycle in one engine body ?

CHAPTER 2

Variable Cycle Engine Philosophy

2.1 Introduction

The concept of "Variable Geometry" may have an extensive as well as restrictive signification. We must therefore first suitably define this concept. An aero gas turbine engine is said to have variable geometry if the off-design behaviour can be influenced by exceeding the limit set by the RPMs of the controlled rotor. We should make the difference between an engine which uses variable geometry and a variable cycle engine.

Variable geometry is used extensively in advanced aircraft engines. In most subsonic commercial aircraft, the engines have variable stators in the intermediate pressure compressor (or booster in the case of two spool engines) and high pressure compressor. In the military field, variable inlet guide vanes are commonly employed, as well as variable stators. In the supersonic aircraft, where reheat is used, the variable exit nozzle is used. This variability in the propelling nozzle is also employed to improve the off-design performance of the engine. In this case, the focus of variable geometry is engine handling only.

However with these techniques it is still not possible to satisfy all the requirements for advanced future supersonic civil transport aircraft. Hence the necessity of a variable cycle engine. A VCE can be defined as one that operates with two or more thermodynamic cycles, being a possible solution to reconciling the necessary, but some times contradictory, performance at different operating conditions. In our case the

VCE will operate as a low bypass ratio turbofan at supersonic cruise and as a medium bypass ratio at subsonic and take-off points.

In order to achieve this change in the operation, the VCE can employ variable geometry in the compressors, turbines and nozzles, as well as the possibility of reheat in the core and/or bypass streams, and variable area mixers. These features should be clearly understood before going deeper into these cycles. The control system will play an important role in future VCE design.

2.2 Engine With Variable Geometry Components

A conventional civil aero engine has one design point where it operates most of the time, the component efficiencies are maximum at this design point and the engine is said to be optimised at this point. However at other off-design points (take-off, climb, descent) the component efficiencies deteriorate and the engine suffers in term of fuel consumption. Some components may reach the limit of its operational range at one or more of these off-design points and the engine thrust is restricted. In order to solve this problem, variable geometry in one or more components may be used.

Figure 2.2 shows a visionary engine (Ref. 23). The purpose of the disclosed difference between the upper and the lower engine halves is to point out possible localities for variable geometry. The numbers are arranged so that they give the chronological order of the engine sections passed by the gasflow. In other words, the lowest numbers represent the inlet diffuser, whereas the highest numbers represent reheating and the outlet section. The left side of the table in Figure 2.2 shows those sections (localities) with variable geometry that are todays practice, whereas the right side shows those that have been tested in prototypes, discussed in technical bureaus and have been applied for patent.

The use of the above variable geometry is not intended to change the thermodynamic cycle of the engine but simply to improve the engine performance and handling at off-design witch lasts a relatively short time in comparison with the time spent at the design point.

2.3 Engine With Two Design Points

For future supersonic transport, the aircraft is expected to spend some time cruising at supersonic speed over the sea, and the necessity of avoiding sonic boom over populated areas may require a portion of the flight to be conducted at subsonic cruise speeds. Moreover the engine noise at take-off should meet the environment regulations. The time ratio between supersonic and subsonic cruise is close to one and depends on the destination routes. From this, we can identify three design points to be considered for the VCE:

a- Supersonic point
b- Subsonic point
c- Take-off

Now the question to be answered is, which one of these points is the most important for designing a VCE, in other words, at which one of these points the engine component, (intake, compressors, turbines....), should be designed. The engine size will depend directly on this choice.

At take-off the engine will spend only a few minutes and the fuel consumption at this point is not so important, but the nozzle exit jet velocity should be around 400 m/sec to keep the noise level down. On the other hand, the engine will spend a considerable time at subsonic and supersonic points and the fuel consumption is an important factor as well as the size of the engine which will influence the nacelle drag at these points.

The supersonic point has a special importance, for some routes, for example LosAngeles-Sydney, the aircraft will cruise at supersonic speed for the most of the flight envelope, and the subsonic point has no significance. Furthermore, the size of the engine at this point is particularly important because of the shock-wave related drag. For these reasons, the supersonic point is considered to be the most important point for designing VCE.

2.3.1 Take-Off

One of the major objectives of VCE is to meet FAR Part 36 noise level at take-off where low noise and small take-off noise footprint area are required. Studies showed that this aim is achieved when the nozzle exit jet velocity is around 400 m/sec at take-off.

From this we can calculate the inlet mass flow necessary to obtain the net thrust required at this point:

Net thrust required at take-off	181,500 Newton per Engine
Nozzle exit jet velocity	400 m/sec
Inlet mass flow Required	W= 181,500 / 400 = 453.8 kg/sec
LP compressor NDMF ($m.\sqrt{T}/P$)	= 0.07602

This will define the first point of the running line on the LP compressor map. This point is only a function of the net thrust required at take-off which depends on the initial aircraft total weight.

This point will give some kind of identification of the performance of the engine at the supersonic point. In fact, the LP compressor NDMF at supersonic mode could not be totally independent of the one at take-off, because the gap between them is a key issue in the feasibility of variable geometry in the LP compressor. This gap depends mainly on the bypass ratio at the supersonic mode. In other words, the take-off point represents one of the extreme values on the LP compressor running line, the supersonic point represents the other extreme value. The gap between these values is mainly a function of the flight conditions and the bypass ratio at the supersonic mode.

2.3.2 Supersonic Point Optimisation

Knowing the altitude, Mach number and the thrust required at this point, the cycle can be optimised in the same way as a conventional engine. The parameters which have to be selected are: TET, OPR and the bypass ratio, their optimum combination should give the best engine in terms of fuel consumption and specific thrust. The choice of these parameters will determine the HP turbine NDMF. A detailed study of the cycle optimisation can be found in Chapter 3.

The choice of bypass ratio has an important role in the matching procedure between the cycles. A turbojet engine (bypass ratio of zero) gives the best combination between size and SFC, but, as we will see later, it is impossible to match this cycle to the subsonic one. A low bypass ratio will lead to a low inlet mass flow and the LP compressor NDMF, at this point, will move to the left on the LP compressor map increasing the gap with the take-off point. A higher bypass ratio will have an inverse effect moving this point to the right decreasing the gap with the take-off point. Figure 2.1 shows the effect of supersonic bypass ratio on the gap between the LP compressor

NDMFs at supersonic and take-off points. The ratio between these two values could reach 3 when the bypass ratio at supersonic point approaches zero. Furthermore, decreasing the bypass ratio will reduce the LP compressor inlet diameter needed to pass the required mass flow and the mismatch with the take-off diameter is higher. A bypass ratio of around 0.7 is selected for further studies.

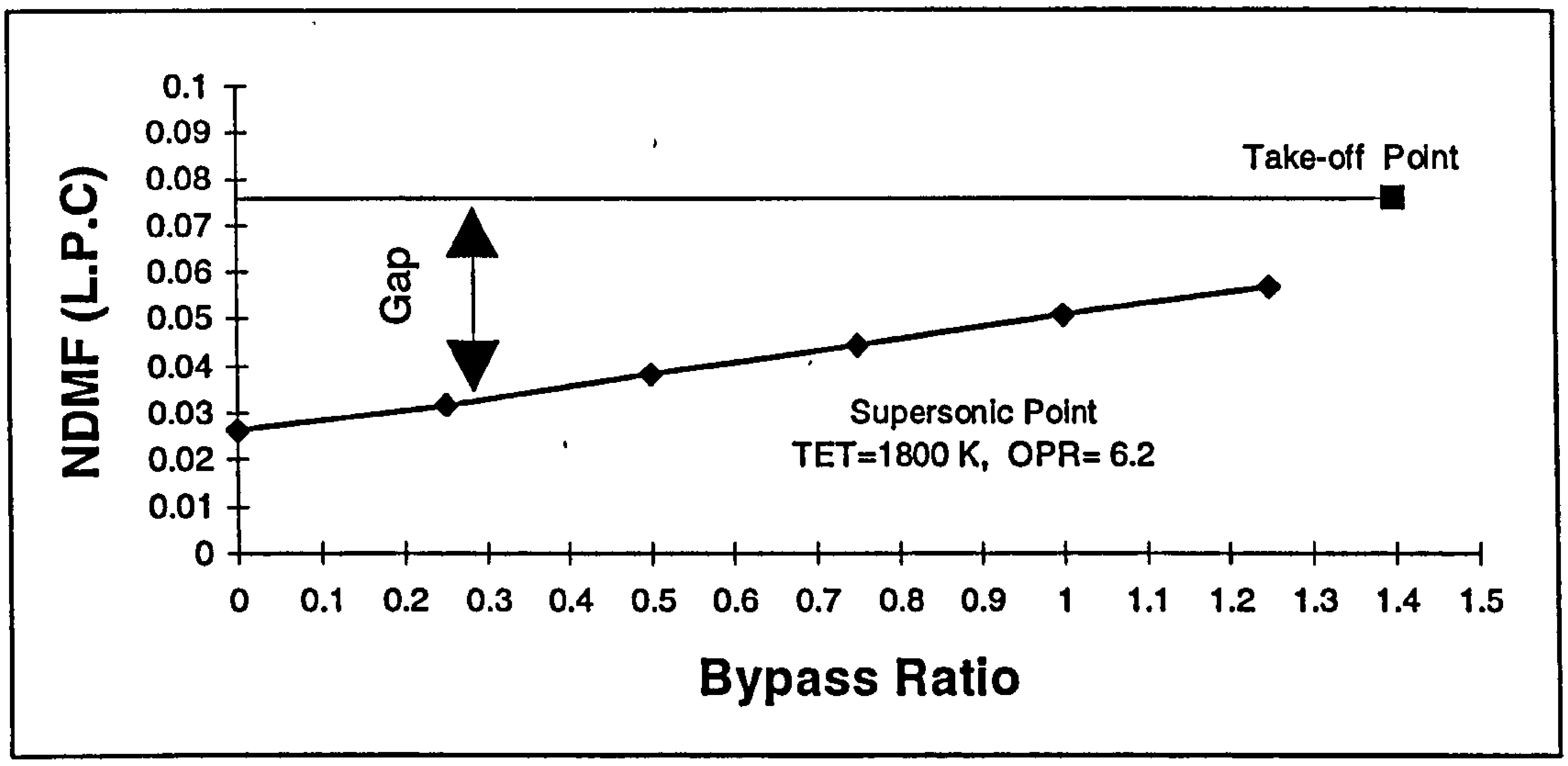


Figure 2.1 Effect Of Supersonic Bypass Ratio On LP Compressor Matching

The choice of the HP compressor pressure ratio, at this point, will determine the LP turbine NDMF. The choice of this pressure ratio depends on the engine configuration. For the TFTJ and MTF engines, the HP compressor pressure ratio has a range of choices because the bypass stream is split after the LP compressor so the IP and HP compressors pressure ratios can be selected in different combinations which all give the same overall pressure ratio, the effect of these combinations on SFCs is very small. For the DBE, the HP compressor pressure ratio is selected in order to match the mixing total pressures of the bypass and the core airflows, the bypass stream is split after the IP compressor unlike the first two engines, so there is only one choice for the HP compressor pressure ratio which gives the best SFC, this will have some consequences on the matching with the subsonic point as it will be seen later.

2.3.3 Subsonic Point Optimisation

This point is an intermediate one between the take-off and the supersonic points. The aim of the optimisation is to improve the installed subsonic SFC, this could be done by improving the cycle performance at this point, mainly, by increasing the bypass ratio, and this is another main objective of the VCE.

The best engine for this mode is a turbofan with a high bypass ratio, but high bypass ratio will lead to difficulties in matching this cycle with the supersonic one, these difficulties consist of matching the turbines and the gap between the points on the LP compressor map. Therefore, a medium bypass ratio is adopted for this mode.

The bypass ratio is the key issue of this optimisation, once it is selected, TET and OPR are determined in such a way that the net thrust required at this point is obtained.

2.4- Matching Two Cycles In One Engine

The future VCE combines two, or more cycles, within the same engine body. The cycle matching is the most crucial matter in VCE because component design requirements depend on cycle data. A survey of the literature has revealed that little research has been published out in this field. An advanced, reported study in cycle matching was carried out by Williamson (Ref. 44) and Nascimento (Ref. 10).

Williamson's method is based on making sure that the non-dimensional mass flow of each turbine is the same in the two operating modes, assuming fixed geometry turbines is used. That is, the turbine is assumed to be choked at the design points for each mode. Hence the size of the engine is determined by turbine components alone. This then ensures that variable geometry is not necessary in the turbine but automatically introduced for the compressors. Nascimento's method enables the designer to match the cycles and to improve compressor operating conditions without affecting engine optimisation.

2.4.1 Matching Philosophy

In the following paragraph two cycles are supposed to be matched. The first one is to be optimised at the supersonic design point which defined by the altitude, Mach number and the net thrust. This cycle is called the HP mode. The second cycle is to be optimised at the subsonic point which called the LP mode.

supersonic design point			**subsonic design point**		
	HP Mode			LP Mode	
Altitude	Mach number	Net thrust	Altitude	Mach number	Net thrust
18,750 m	2.7	82,500 N	9,150 m	0.95	56,750 N

The cycle optimisations will give two different engines each one is optimised at its design point. These engines will have different sizes and different component characteristics, and the question know is whether if we can manufacture one engine which is capable of yielding the same performance as the above engines at the two design points.

The answer to this question depends on whether we can match the different engine components in the two cycles, in other words, each component in the new engine should be able to give the same kind of results as the two separate components in the two engines. These components include turbines, compressors, nozzle areas and mixing area.

Studies showed that matching is possible, but a penalty was to be paid in terms of cycles optimisation. In fact, one or the two cycles have to move away from the optimised point in order to be able to match them and the loss in performance is to be accounted.

2.4.2 The Engine Matching Procedure

This procedure consists first of matching the turbines, and then matching the compressors, nozzle and the mixing area. This matching procedure could be divided into two main stages.

First Stage:

In the first stage the two or more cycles are treated as independent engines, the major parameters of the various cycles are assessed and analysed. From the point of view of cycle selection, high bypass ratios and low fan pressure ratios are suitable for high subsonic speed applications. As the flight speed increases, the requirement is for lower bypass ratios and higher fan pressure ratios. The cycle parameters are selected in such a way that we will have, at the end of this stage, two separate engines with their turbines matched. The next steps are to be followed:

A- Optimisation of the supersonic cycle. This will give the best combination of TET, OPR and bypass ratio in terms of SFC and specific thrust. This will give the best engine for the supersonic design point.

B- Optimisation of the subsonic cycle. This will lead to the best combination of TET, OPR and bypass ratio in terms of SFC and specific thrust. This will give the best engine for the subsonic design point.

C- Check if the turbines of the first engine, at the supersonic mode, are matched with the turbines of the second engine, at subsonic mode, (usually they are not). If not, change the subsonic cycle parameters, (BPR, OPR, TET) until the turbines are matched with the supersonic mode. Some times, the supersonic cycle parameters need to be slightly changed.

In this work, the supersonic point is considered the most important point, therefore its optimised cycle parameters are very little changed, most of the changes take place in the subsonic cycle parameters. At the end of this stage, we should have two different engines with their turbines matched, (at least one of them is not optimised). These engines will have different nozzle and mixing areas and different compressor characteristics. These parameters will be very useful for the next stage in the engine matching.

Second Stage:

This stage consists of studing the behaviour of the VCE as one engine which is supposed to have the same performance as the two above engines. This includes generating the compressor operating lines and check if they can be followed by a single compressor, with or without variable geometry. The transition from one mode to other is carried out and the nozzle and mixing areas are checked against the areas found in the first stage. This stage is carried out by using an updated version of Turbomatch program. The link between the first stage and the second one are the following parameters:

- The turbines non-dimensional mass flow of the engines in the first stage are the same as the VCE.
- The VCE nozzle throat and exit areas should be the same as the areas found in the first stage corresponding to the two design points.
- The same thing applies to the mixing area.

It is worth mentioning that the first stage in the matching procedure is used as a reference point and it was used in the past because Turbomatch, at that time, was not able to simulate VCEs, therefore it was necessary to simulate the two engines separately. With the new version of Turbomatch the turbines matching is done automatically by the program and the performance at the subsonic point is considered as an off-design calculation. However, the first stage in the matching procedure is still very important to understand the behaviour of VCE through out the flight envelope, although it is a time consuming procedure. Users with some experience of VCEs can use Turbomatch directly without going through steps B and C of the first stage of the matching procedure.

2.4.3 Turbine Matching

Turbine matching consists of ensuring equality of non-dimensional mass flow of the turbines at the HP and LP modes. This process of ensuring the equality of non-dimensional mass flow is called matching. In the case of the two spool engine, the HP turbine will be matched first and then the LP turbine.

2.4.3.1 Matching Of The HP Turbine

The value of the HP turbine non-dimensional mass flow is fully defined when the engine was optimised at the supersonic design point. This value is only a function of the main cycle parameters (TET, OPR and BPR) at this point. As previously mentioned, these parameters will not be changed for cycles matching, and the engine will be optimised at this point. On the other hand, the subsonic optimised cycle parameters are to be changed in order to obtain the same above value of the HP turbine non-dimensional mass flow.

Figure 2.3 shows the effect of subsonic cycle parameters (TET, OPR and BPR) on the HP turbine non-dimensional mass flow for the TFTJ engine. On this chart, we can see the HP turbine NDMF at the supersonic mode represented by the horizontal line, and three groups of curves. Each group represents the variation of HP turbine NDMF in function of overall pressure ratio for one bypass ratio and three different TETs. These curves are plotted for an inlet total pressure of 0.53 atm and constant inlet total mass flow of 250 kg/sec (this value is limited by the compressor matching). The intersection of the horizontal line with theses curves represents all the subsonic cycles witch have their HP turbine matched with the supersonic cycle. It should be noted that the exact net thrust required at this point is not obtained by all these points. Points A, B, and C give the exact net thrust required as well as their HP turbines are matched. Each one of these points represnts a solution to the HP turbine matching problem.

Point A represents a turbofan with a relatively high bypass ratio of 2.3, relatively low overall pressure ratio of 6.56 and a TET of 1300 K, it has the highest bypass ratio and lowest overall pressure ratio. Point C is an intermediate point with a bypass ratio of 1.8, an overall pressure ratio of 7.4 and a TET of 1200 K. Point B represents a turbofan with relatively low bypass ratio of 1.3, relatively high overall pressure ratio of 8.85 and a TET of 1100 K, this point has the lowest bypass ratio and highest overall pressure ratio. The TETs are relatively low because the core massflow is high, decreasing the inlet total massflow, i.e. increasing the specific thrust, will lead to higher TETs and lower overall pressure ratios in order to maintain the HP turbine NDMF constant. More details about the SFC and the specific thrust of these engines can be found in Chapter 3.

Increasing the subsonic bypass ratio is coupled with decrease in overall pressure ratio (Fig. 2.3), this means that the advantage gained in increasing the bypass ratio would be lost by the decrease in overall pressure ratio. The effect of TET is less important than the bypass ratio and OPR, this due to the fact that the square root of TET is counted in the calculation of the NDMF. Simulation studies done by Turbomatch program showed that point C gives the best SFC (4% lower than point A), this is because the overall pressure ratio decreases SFC faster than the increase due to the bypass ratio.

The matching of HP turbine will define the main cycles parameters (TET, OPR and BPR) for both modes. Only the distribution of the pressure ratios on the LP, IP and HP compressor is left for matching the LP turbine.

2.4.3.2 Matching Of The LP Turbine

The value of the LP turbine non-dimensional mass flow is a function of the main cycle parameters (TET, OPR and BPR) and the HP compressor pressure ratio. Figure 2.4 shows the variation of the LP turbine NDMF in function of the HP compressor pressure ratio at the supersonic mode. Increasing the HP compressor pressure ratio will increase the LP turbine NDMF, this is because the HP turbine work is increased in order to match the increase in the HP compressor work due to the increase in its pressure ratio and this would lead to a lower HP turbine exit total pressure. A lower inlet pressure at the LP turbine would increase its NDMF.

Figure 2.5 shows the position of the LP turbine NDMF at the supersonic mode represented by the horizontal line and three curves corresponding to the three above points A, B, and C where the HP turbines are matched and the net thrust required is obtained. Each curve shows the variation of the LP turbine NDMF in function of the HP compressor pressure ratio at the subsonic mode. The intersection points between the horizontal line and these curves would give the subsonic cycles which have their LP turbines matched as well as their HP turbines. These intersection points define the HP compressor pressure ratios at the subsonic mode.

Figures 2.6, 2.7 and 2.8 are used to define the pressure ratios for the LP and IP compressors at the subsonic mode for points A, B and C respectively. In fact, the overall pressure ratio and the HP compressor pressure ratio are known, the pressure ratios for the LP and IP compressors are defined in such a way that the mixing total pressure is the same for the bypass stream and the core stream. The above figures show that there is only one solution for each point.

On Figure 2.6 (point A , high bypass ratio and low overall pressure ratio), the IP compressor pressure ratio is 1.2 (which is the minimum accepted for designing compressors). If the LP turbine NDMF at the supersonic mode increase (Figure 2.4) the HP compressor pressure ratio would increase and there will be no solution to the problem for point A. Therefore, the choice of the LP turbine NDMF at the supersonic mode has an important role in selecting the compressor pressure ratios at the subsonic mode. Increasing this value will tend to favour lower bypass ratio and higher overall pressure ratio cycles at the subsonic mode, and this will lead to a lower HP compressor pressure ratio and higher IP compressor pressure ratio.

The above procedure as described treats fixed geometrry turbines as mentioned earlier. The inclusion of turbine variable geometry is very simple. The steps of turbine matching are modified to allow for differences in turbine non-dimensional mass flow, these differences reflecting the variability of turbine nozzle area (see Chapter 6).

2.4.4 Compressor Matching

The next part of the matching procedure is to ascertain that the design of the compressors can satisfy the requirements imposed by the turbines in the two cycles. This method of compressor matching also consists of two steps: the determination of the compressor running lines using Turbomatch program and the preliminary analysis of the compressors. The HP turbine matching determines the overall pressure ratio while the LP turbine matching determines the LP, IP and HP compressor pressure ratios in both modes. The choices of these pressure ratios can be limited by the LP turbine matching between the two cycles. Using variable geometry in the LP turbine would increase the range of these choices (see Chapter 6).

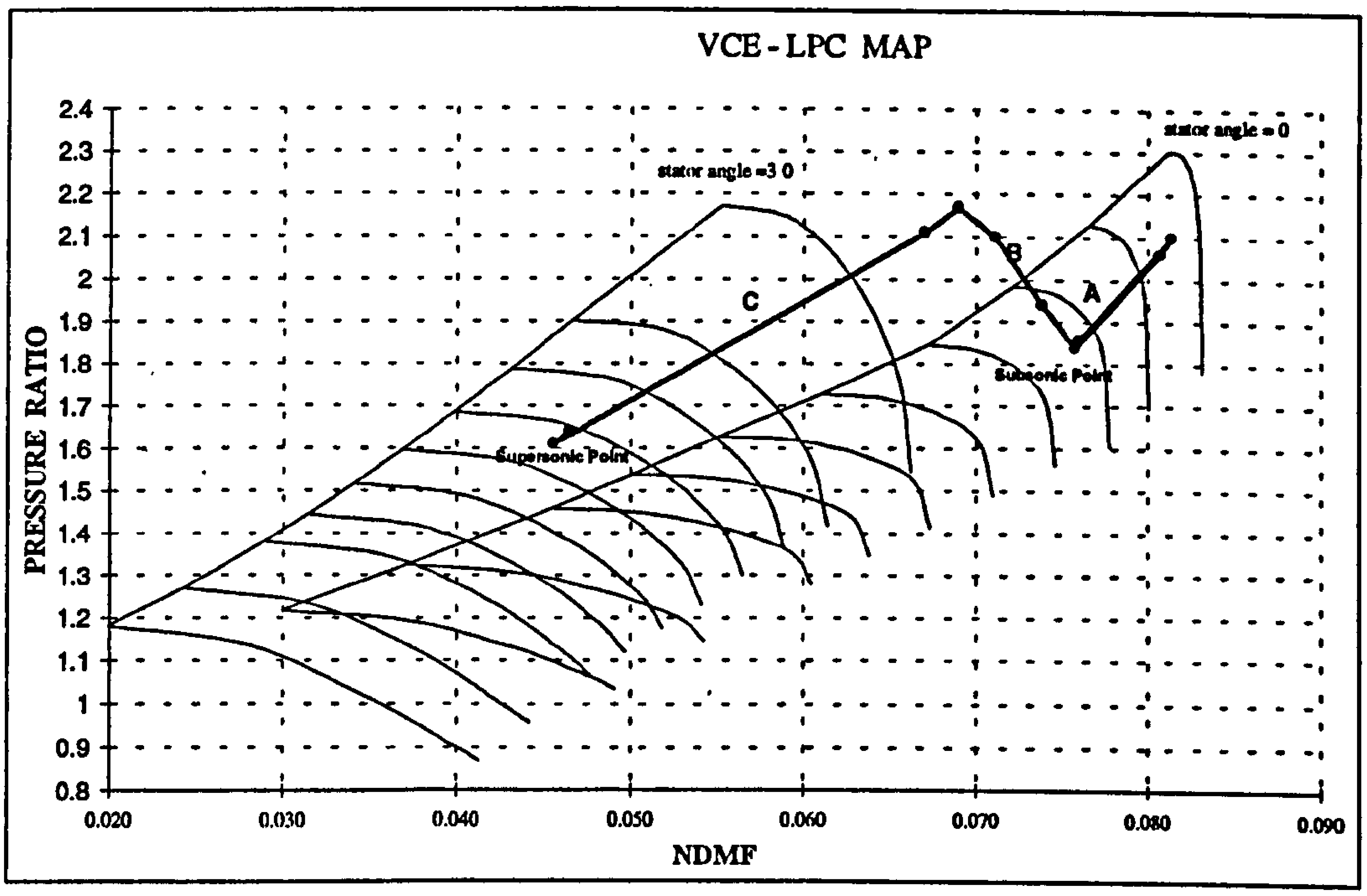


Figure 2.9 Example Of A VCE LP compressor running line.

Compressor Running Lines

The compressor running lines are obtained by using Turbomatch off-design facilities throughout the flight envelope. Two aims are targeted at each off-design point, the first is that the net thrust required at this point should be obtained, the second is to make the changes necessary to transform the cycle from one mode to other. The parameters used for this purpose are:

TET
LP and IP compressor variable stators
Nozzle throat and exit areas
Mixing areas (area just behind the LP turbine exit)

Once these targets are met, compressor data are taken from Turbomatch output file and plotted using Excel program. Each running line has the three different following parts:

A- First part: from take-off to the end of subsonic cruise, where the cycle is fixed and only TET and nozzle areas are changed in order to obtain the required net thrust.

B- Second part: during which the cycle is changed progressively from medium bypass ratio to low bypass ratio turbofan.

C- Third part:climb and acceleration from subsonic to supersonic point while the cycle is still changing to the supersonic low bypass ratio required.

The shape of the running lines on the LP and IP compressor maps could be awkward, but it has a conventional shape on the HP compressor map because it has not variable geometry and the turbines supposed to have fixed geometry. Some times this shape could be slightly changed by alternating the above variables.

At the end of this stage, the running lines are obtained and the amount of variable geometry needed for each compressor is assessed. This simulation is done by using the Turbomatch compressor standard maps. It is necessary now to design the compressors and assess their performance.

Compressor Performance Analysis

Having obtained the position of the running lines, the next step is to design the compressor that can operate in the required conditions. Therefore, both design and off-design compressor analysis is required. The design point analysis will produce the compressor geometry, and the off-design analysis will produce the compressor characteristics. These have been carried out in Chapter 4 and it is anticipated that the necessary use of variable geometry will be extensive.

If the performance of the compressors is not satisfactory, the main cycle parameters are reviewed and the calculation repeated. Normally not more than two or three iterations are necessary. If the performance of the compressor is deemed to be satisfactory, the analysis of the engine can now be refined.

2.4.5 Effect Of Supersonic Mach Number On Turbine Matching

A Mach number of 2.7 and altitude of 18,750 m will lead to an inlet total pressure of 1.3 atm. and total temperature of 530 K. These inlet conditions will limit the overall pressure ratio to around 7 because of the limit on the HP compressor exit temperature. This low OPR leads to relatively high turbine NDMF at the supersonic point, and the matching with the subsonic mode will limit the OPR to around 8.5 and the bypass ratio to around 1.5.

Reducing the supersonic Mach number to 2.15 and the altitude to 15,000 m, will lead to an inlet total pressure of 1.07 atm. and inlet total temperature of 417 K. The maximum OPR is now limited to around 15.5 and the turbine NDMF are reduced by over 60%. The matching with the subsonic cycle will be much easier and will increase the OPR and the bypass ratio giving better subsonic performance.

Decreasing the supersonic flight Mach number will increase the journey time and decrease aircraft productivity, but it will increase the feasibility and reliability of the future VCE.

2.5- Three Candidates VCE For Future SSTs

The powerplants for future supersonic civil transport are based on a conventional mixed-flow augmented turbofan engine and then variable geometry features are added to this conventional engine cycle in order to obtain the best performance and range capability (Ref. 14) for the future supersonic airplane.

For the application of a supersonic civil transport, capable of cruising at Mach 2.7 and with reasonable performance at Mach 0.95, three variable cycle engines were selected for examination. They are the following.

Turbofan-Turbojet
Mid-Tandem Fan
Double Bypass Engine.

The three engines are low bypass ratio turbofans at the supersonic mode and medium bypass ratio turbofans at take-off and subsonic mode. They all have two spools with three compressors and two turbines. The LPC, IPC and LPT are on the LP spool while the HPC and HPT are on the HP spool. The LPC and IPC compressors have variable stators. The turbines have a fixed geometry.

In some circumstances the availability of a 'high thrust mode', to be used during climb and perhaps also for take-off, would have attractions particularly if turbine entry temperature increases could be avoided. The proportion of the total engine airflow which passes through the core can be raised by increasing the HP compressor entry temperature and this is the main aim of using the IPC or 'booster' on the LP spool behind the LPC. The IP booster variables are operated in conjunction with a reduction in bypass propelling nozzle area, forcing the cycle to rematch with reduced bypass ratio at a given fan speed (Ref 11).

2.5.1 Turbofan-Turbojet Engine

The Turbofan Turbojet (Fig. 2.9) is a mixed stream turbofan. It has two spools, three compressors, two turbines, a fully variable convergent divergent nozzle and a variable area bypass duct. An after burner is required to increase the thrust at certain points of the mission, such as the transonic acceleration where the thrust required is higher than that provided by the dry engine. The flow is separated into two streams after the LP compressor. The bypass stream is then mixed again with the core stream after the LP turbine.

During maximum power and supersonic operation, the engine is run as a low dual rotor turbofan engine with a high percentage of the LP compressor flow being passed through the core compressor. The bypass duct is almost closed with only a small flow allowed to pass through it. In this case the engine operates with a low bypass ratio, suited to the flight regime. For subsonic flight the outer duct and nozzle throat and LP compressor stators are open to allow the increase in bypass ratio and the cycle is changed from low bypass to a higher one.

2.5.2 Mid-Tandem Fan

The philosophy of the design of the Mid Tandem Fan engine (Fig. 2.10) is to achieve the benefits of a turbofan, but with a smaller inlet diameter. To achieve this, the fan is mounted behind the IP compressor where the diameter is smaller, thus the compressor frontal area is reduced.

In the LP mode (subsonic mode) the bypass flow enters the engine via both the frontal air inlet and the lateral air inlets and mixes again with the main stream after the LP turbine to be discharged by the convergent-divergent nozzle.

In the HP mode (supersonic mode) the auxiliary intakes are closed and the bypass flow is reduced by modulating the mixer and nozzle areas.

2.5.3 Double Bypass Engine

The Double Bypass Engine (Fig. 2.11) has two spools, three compressors, two turbines, a fully variable convergent divergent nozzle and a variable area bypass duct.

An after burner is required to increase the thrust at certain points of the mission, such as the transonic acceleration where the thrust required is higher than that provided by the dry engine.

It has two separate bypass ducts. In the subsonic mode the flow is split after the LP compressor and the bypass stream is discharged by a separate convergent nozzle. For supersonic mode the outer bypass duct is closed and the flow is split now after the IP compressor and the inner duct is open to pass the bypass stream which is mixed again with the main stream after the LP turbine to be discharged through the convergent divergent nozzle.

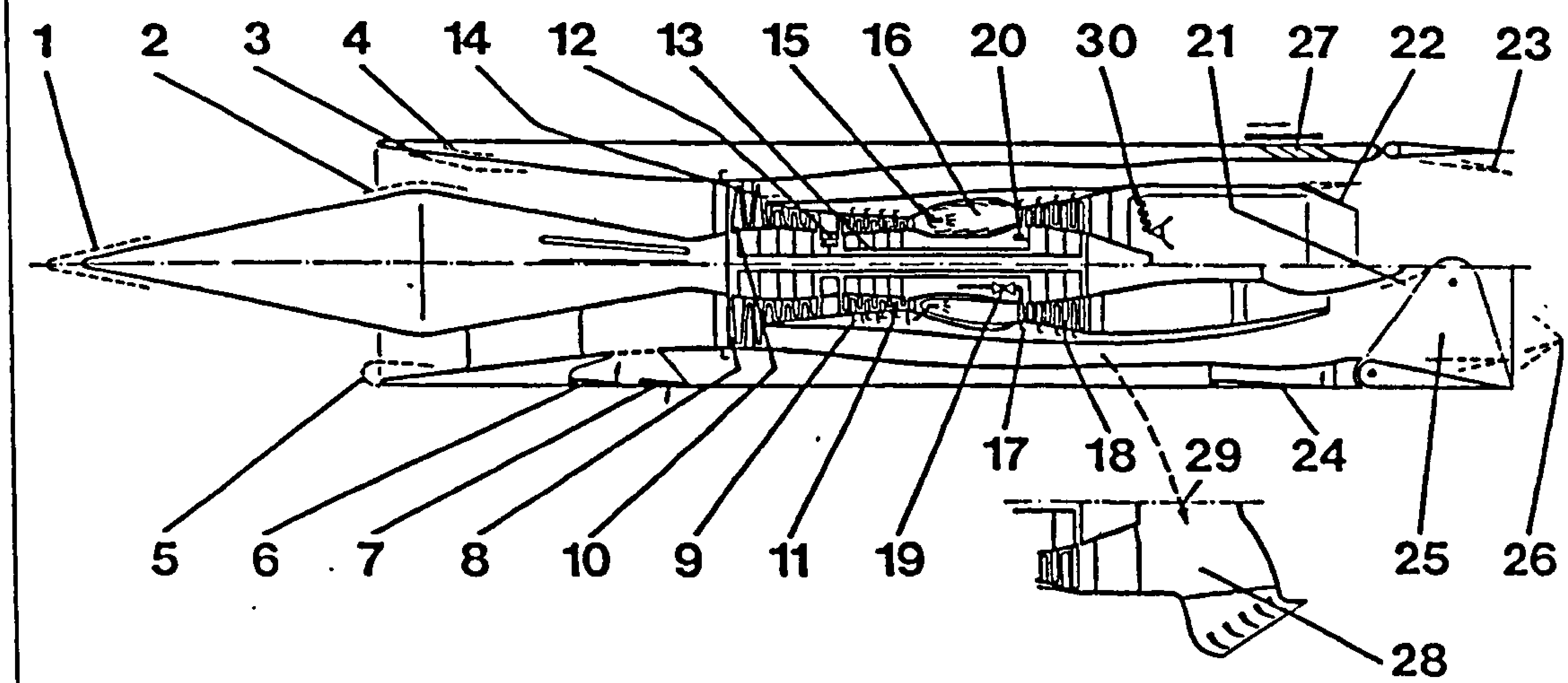
2.6- Advantages And Disadvantages Of VCE

The most important advantage expected from using VCE in future supersonic transport is a substantial range improvements as compared to a conventional engine. These range improvements are mainly achieved by reducing the subsonic SFC by around 15% (relative to a Turboje) and improving the fuel consumption at off-design by the extensive use of variable geometry.

The future VCE will have a low emission combuster and afterburner. The noise level at take-off will be met by FAR part 36 requirement. In other words, the future VCE will be environmentally accepted.

The disadvantages are mainly an increase in the engine weight and a more complex control system, therefore the reliability of the engine will be affected. The performance of any VCE depends critically on the attainment of the predicted technology level improvements.

FIGURE 2.2 Explanations To variable Geometry



	realized in production engines		prototype test / proposal / patent pending etc.
1	Inlet cone displacement in direction of engine main axis	5	Inflatable inlet lip
2	Inlet cone adjustment perpendicular to engine main axis	8	Retractable shutter in front of first compressor blade
3	Angle change of inlet lips	12	Pneumatic or mechanic fit adjustment of labyrinth seals
4	Displacement of inlet lips	14	Splitter adjustment on bypass engine
6	Adjustable auxiliary inlet	16	Combustion chamber with 2 or more primary zones and variable mixture
7	Blow-off between supersonic inlet and engine	17	Turbine with adjustable guide vanes
9	Compressor with adjustable stator	18	Turbine with adjustable rotor blades
10	Compressor with adjustable rotor	19	Throttling or shut-off cooling air channel for hollow blades
11	Controlable blow-off orifices between compressor stages	20	Fit adjustment of labyrinth seals
13	(Free rotating) multispool engine	25	Pneumatic thrust reversal
15	Combustion chamber with switchable injection nozzles (main nozzles - auxiliary nozzles)	27	Shift blade row for thrust reversal
21	Adjustment of minimal thrust nozzle cross-section through shifting of axial core	30	Re-heat with extendible flame holder (mechanically or pneumatic)
22	Adjustment of minimal thrust nozzle cross-section with flaps		
23	Angle change in thrust nozzle expansion section		
24	Rear-end ventilation of expansion section		
25	Mechanical thrust reversal		
26	Shutting and opening the relief channel for thrust reversal		
28	Turnable nozzles for thrust vector control		
29	Dual thrust systems with variable exhaust deflection		

FIGURE 2.3 Effect Of Overall Pressure Ratio On The HPT Non-Dimensional Mass Flow For Different TET And BPR At Subsonic Mode.

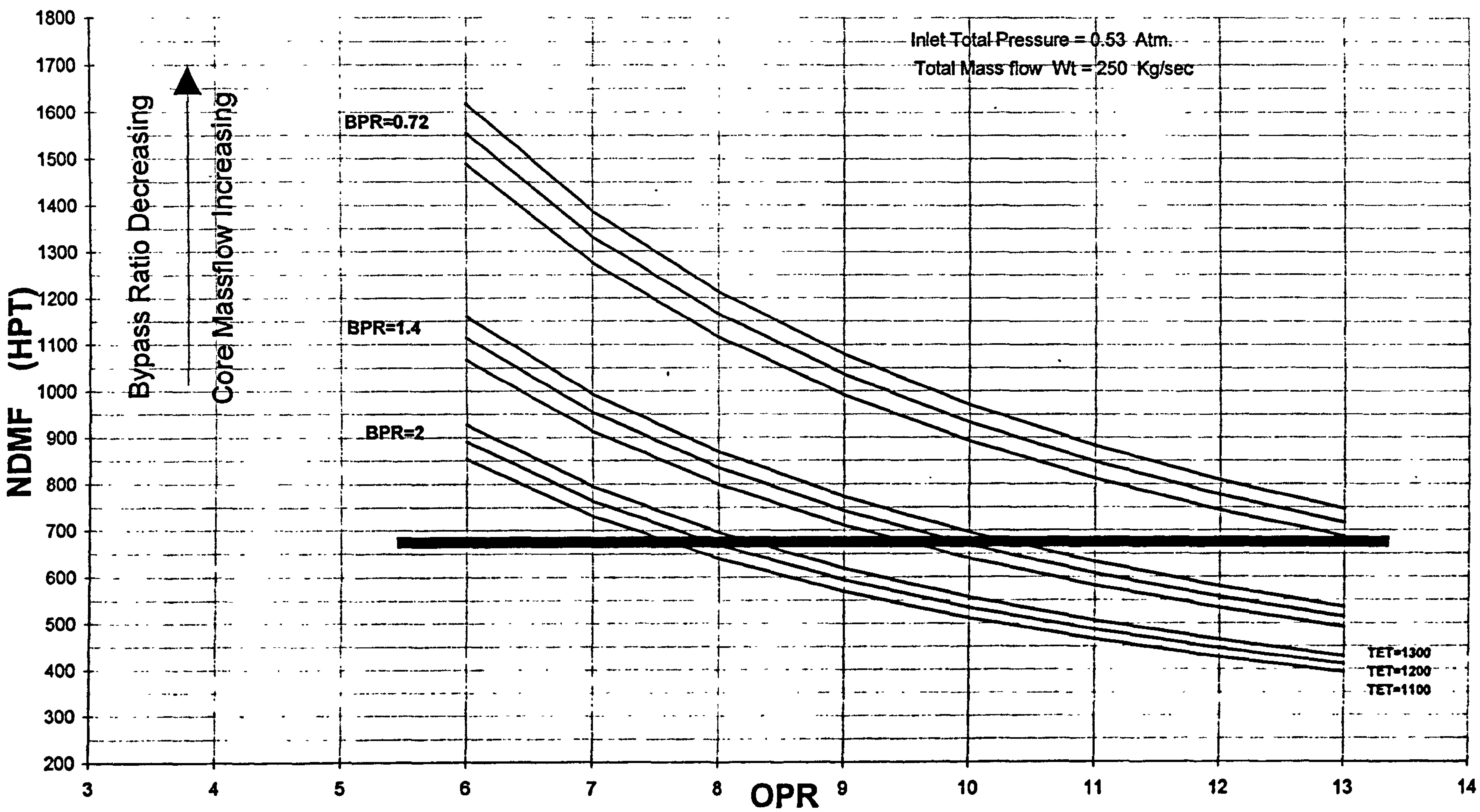


FIGURE 2.4 Effect Of Supersonic HPC Pressur Ratio On LPT NDMF
(TET = 1800 K , OPR= 6.2 , BPR = 0.754)

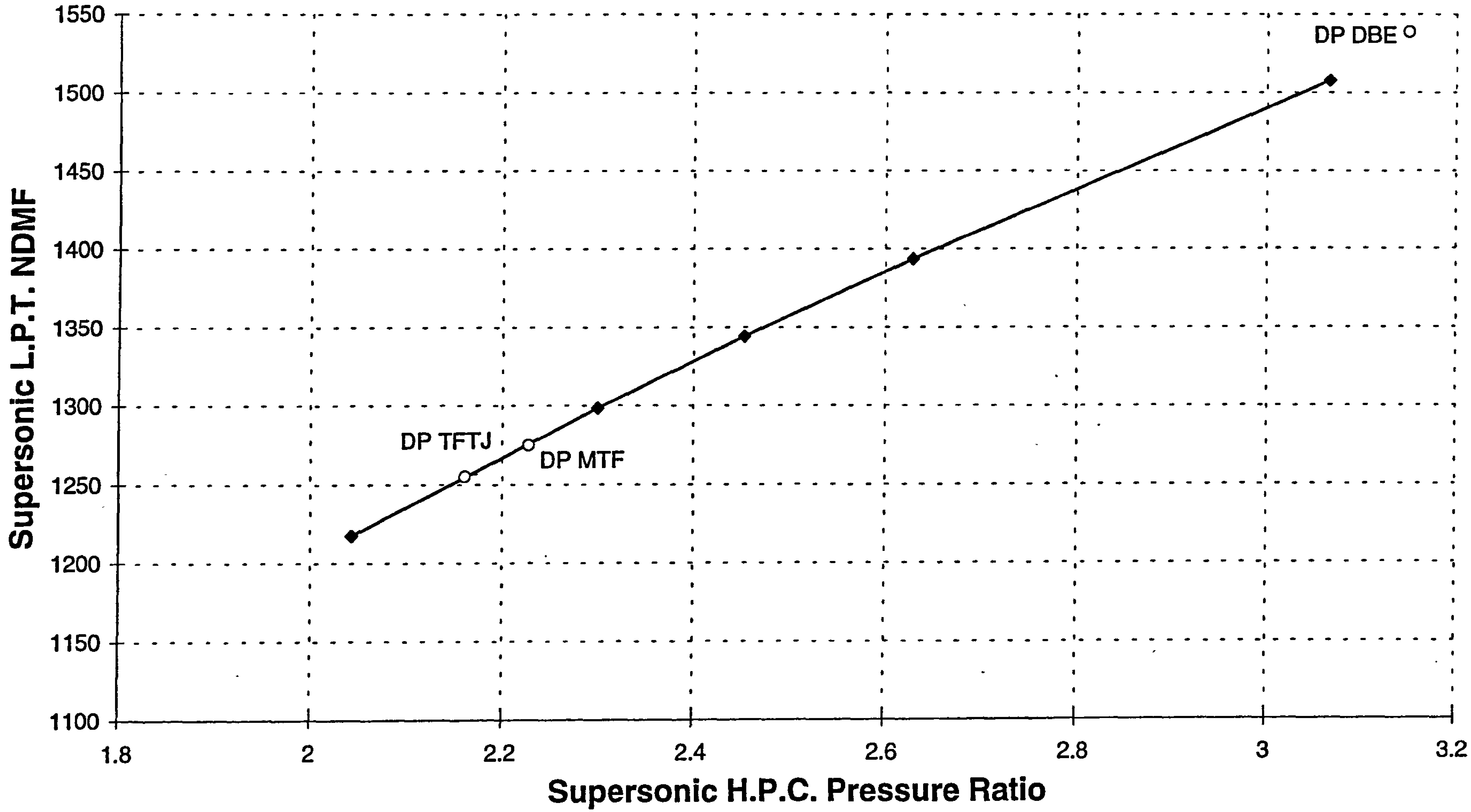


FIGURE 2.5 Effect Of HPC Pressure Ratio On LPT Non-Dimensional Mass Flow For Different Overall Pressure Ratios

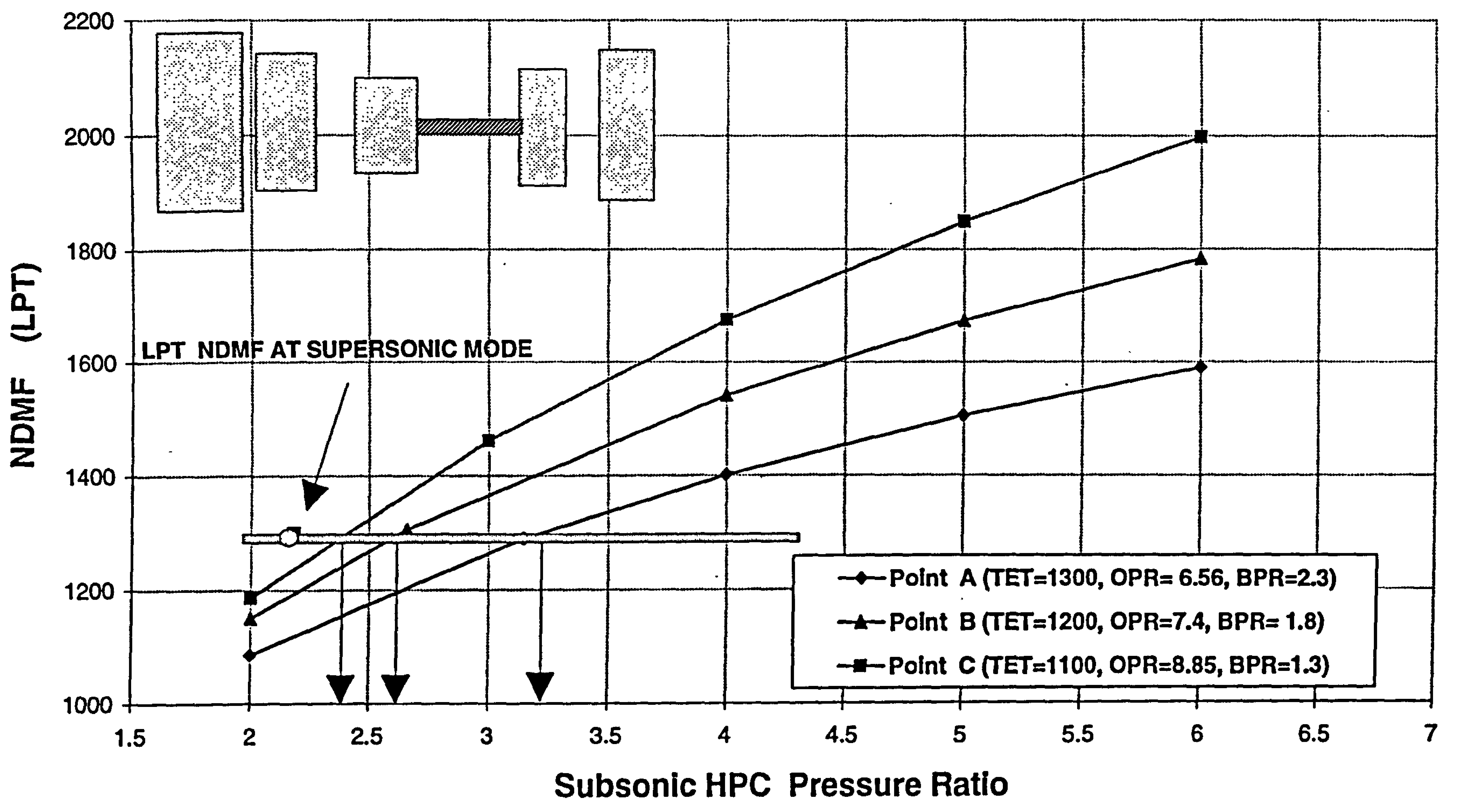


FIGURE 2.6 Effect Of Pressure Ratios On Mixing Total Pressure

POINT A (OPR =6.56 , TET = 1300 , BPR = 2.3)

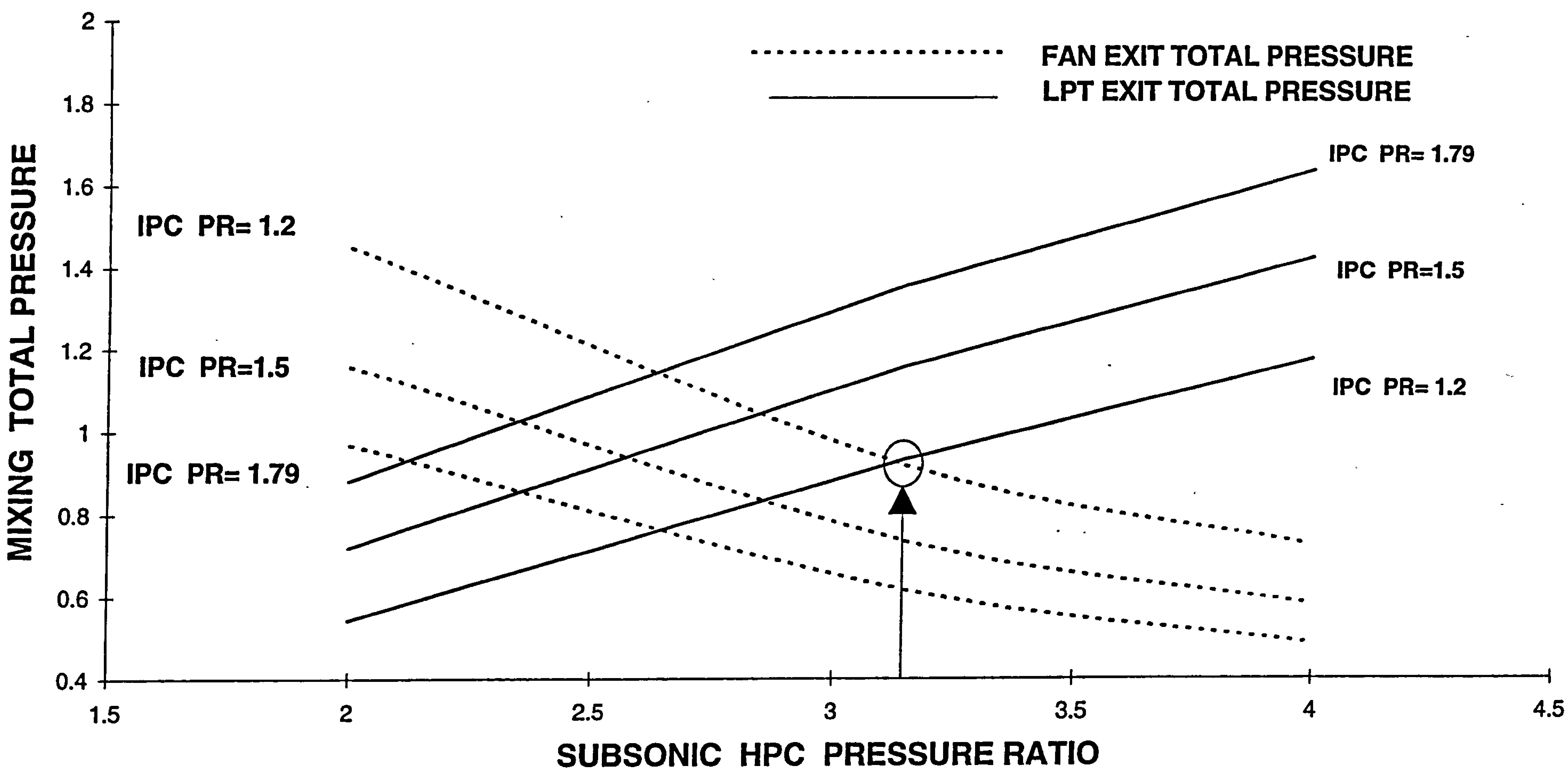


FIGURE 2.7 Effect Of Pressure Ratios On Mixing Total Pressure
Point B (OPR =7.4 , TET = 1200 , BPR = 1.8)

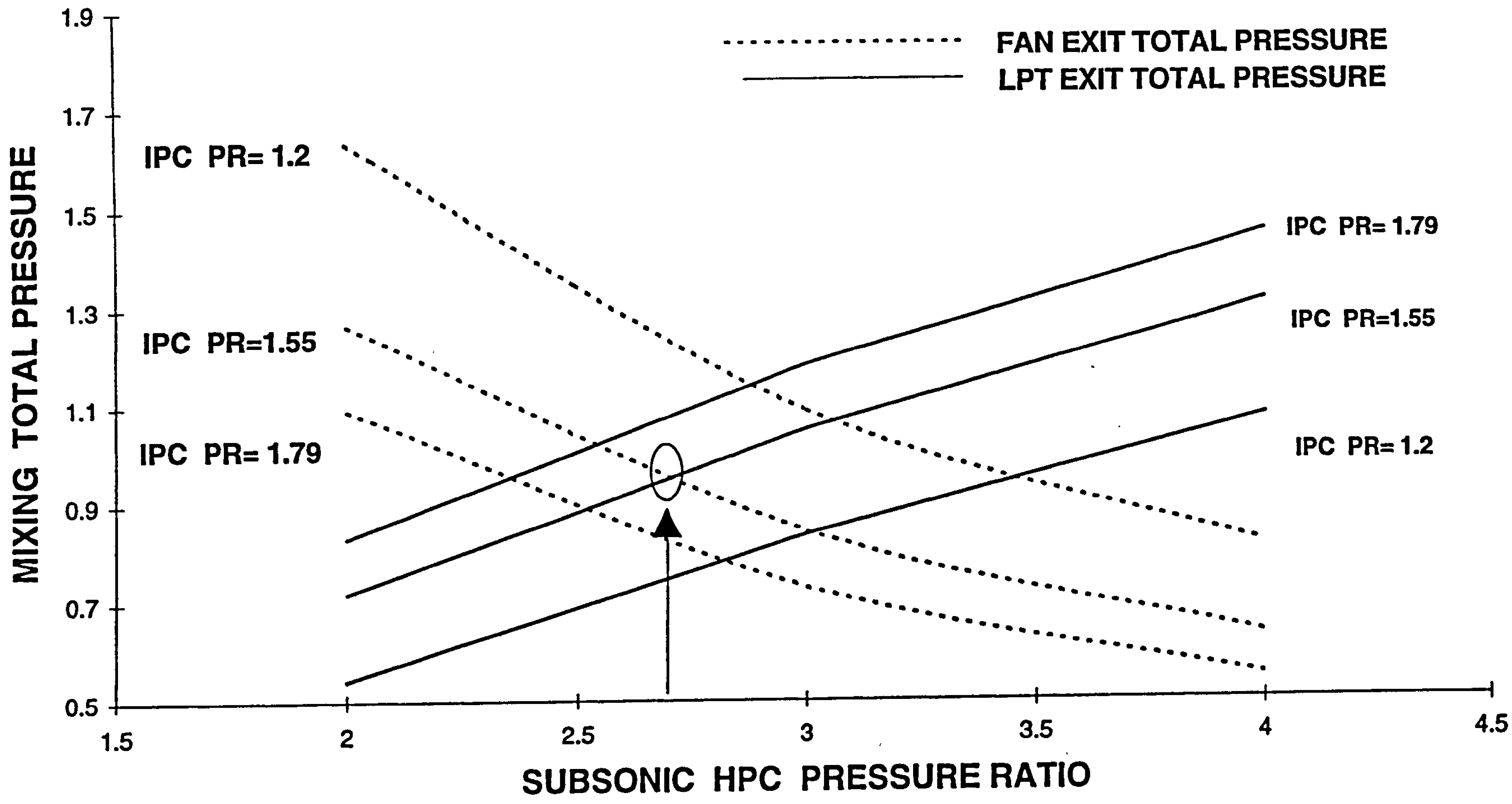


FIGURE 2.8 Effect Of Pressure Ratios On Mixing Total Pressure
Point C (OPR =8.85 , TET = 1100 , BPR = 1.3)

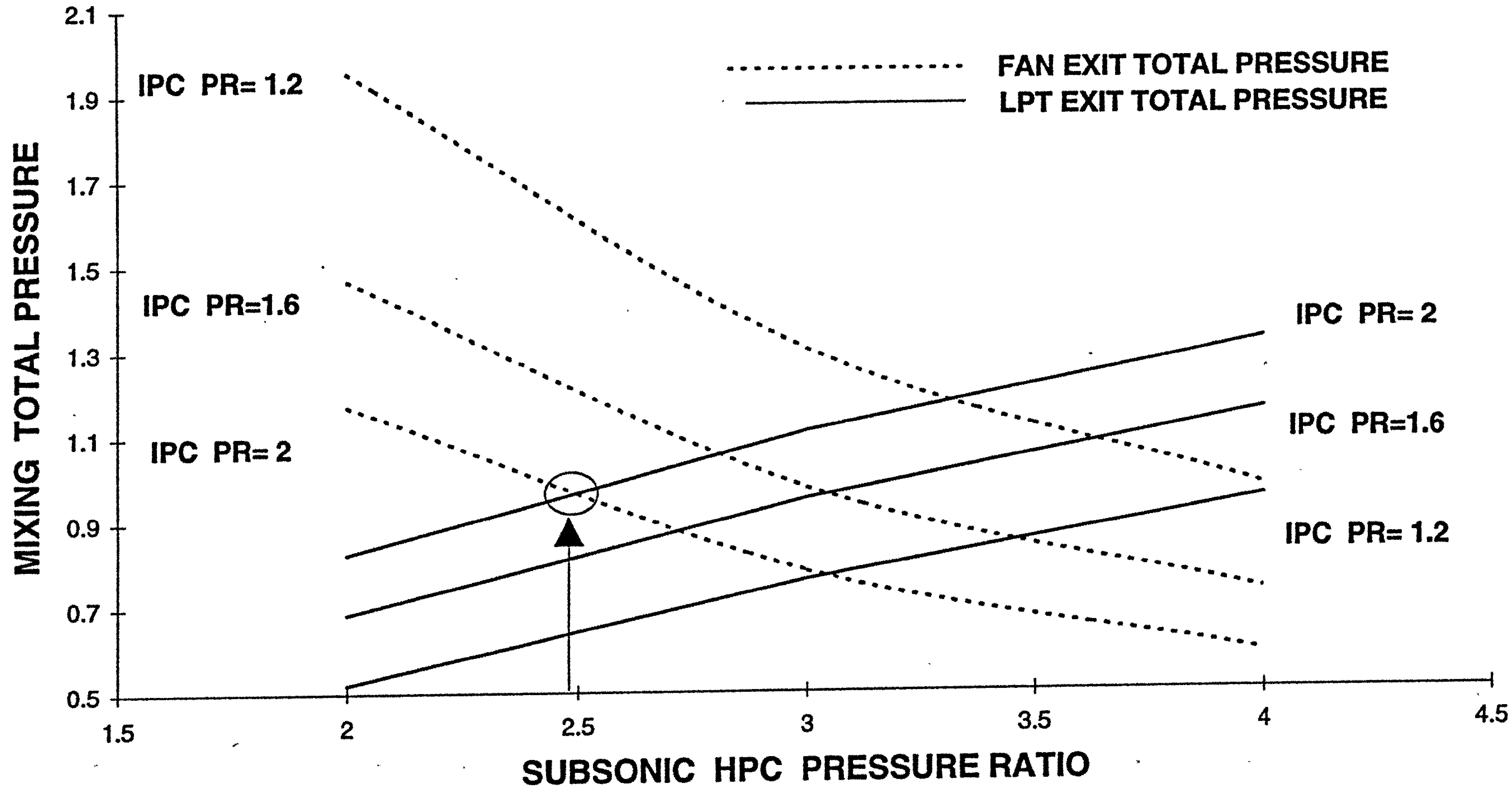
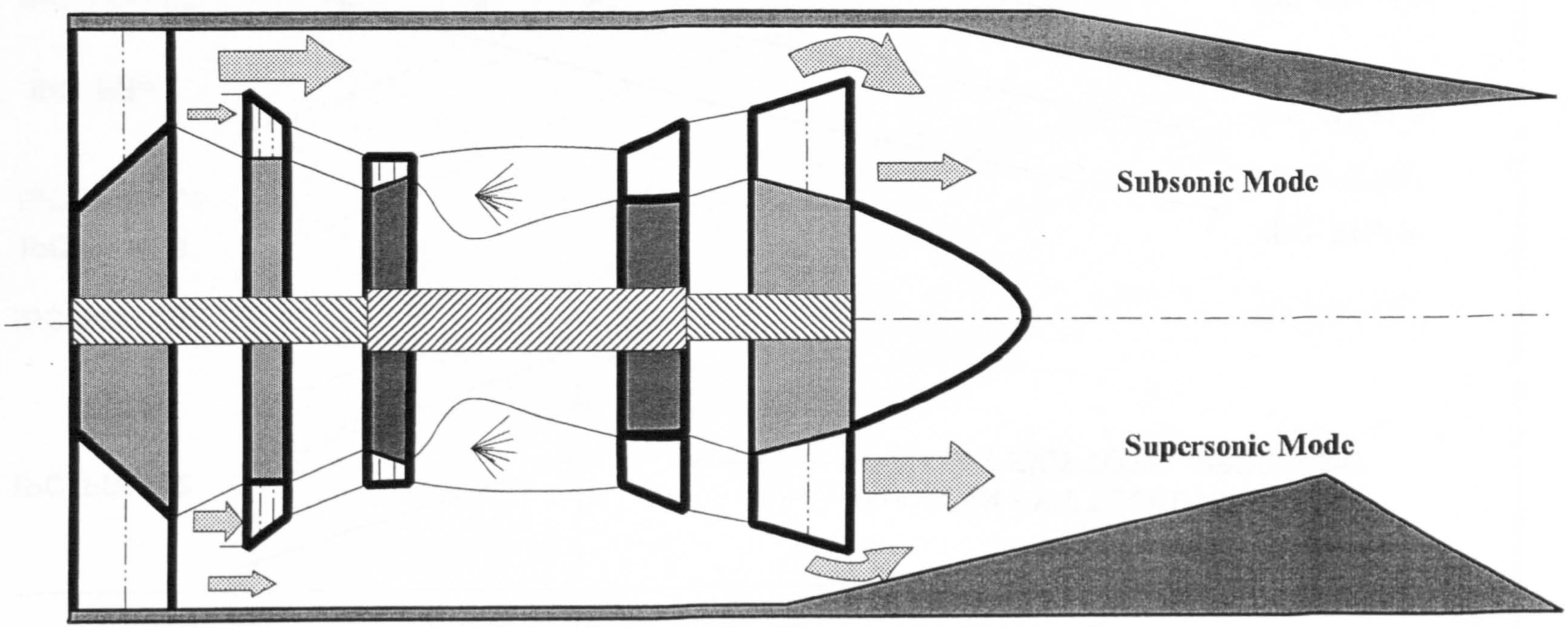


Figure 2.10 *Turbojet Turbofan Engine*



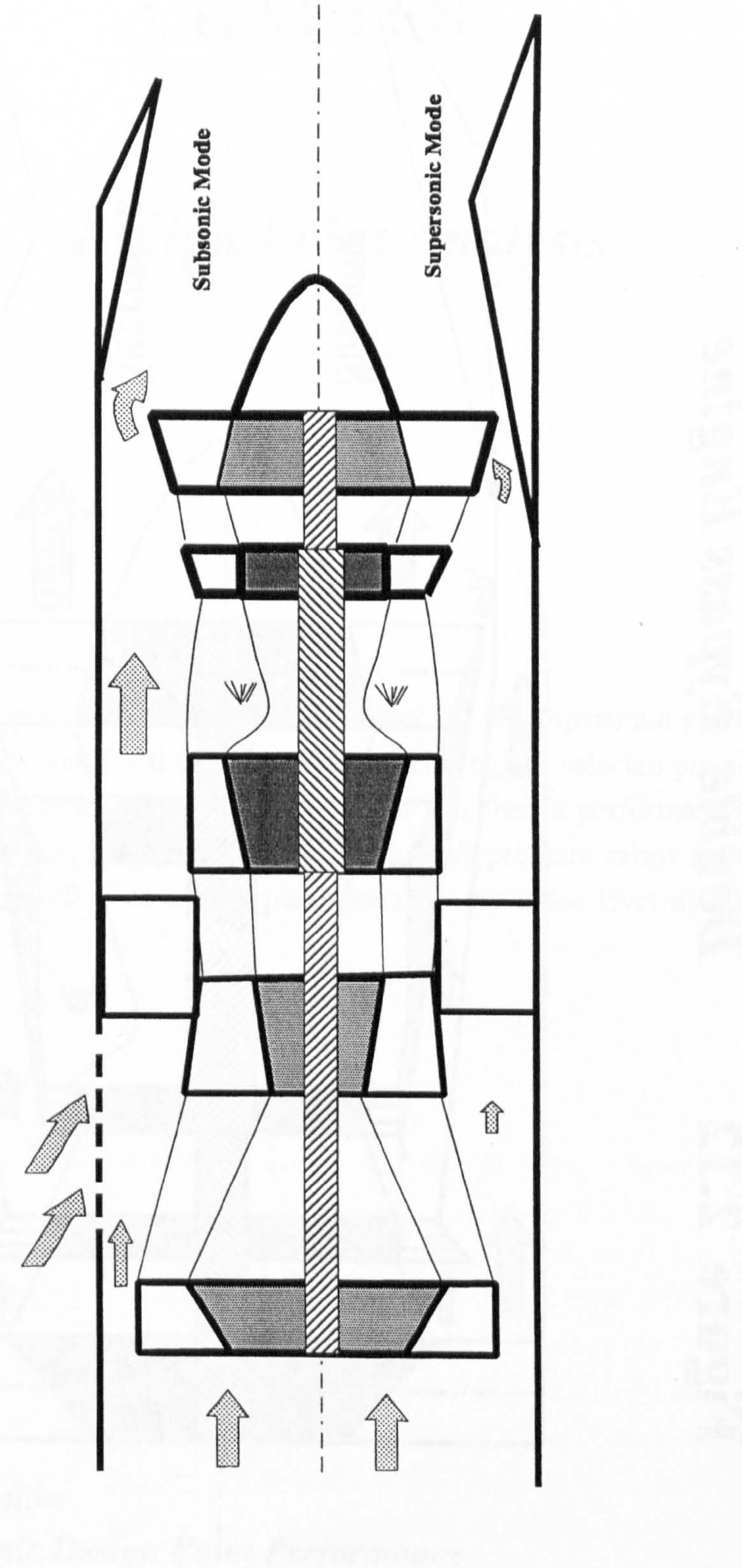
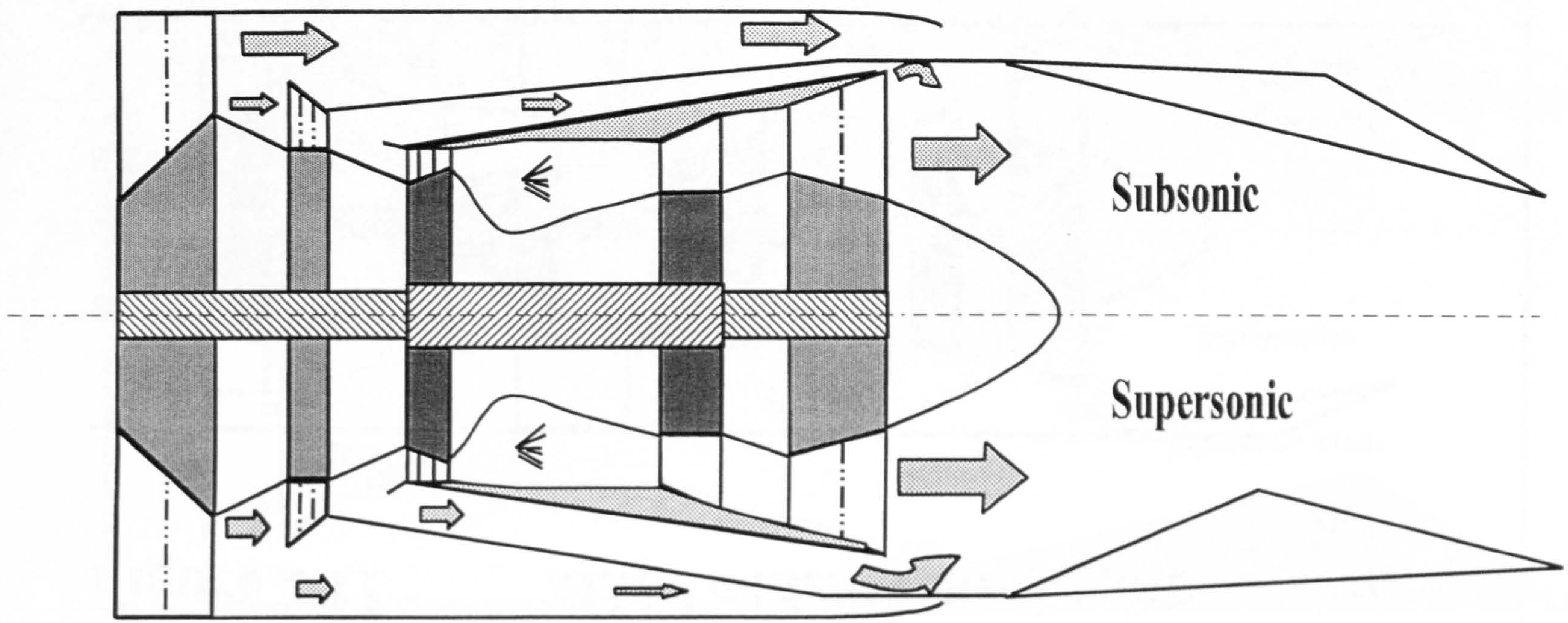


Figure 2.11 *Mid-Tandem Fan Engine*

Figure 2.12 *Double Bypass Engine*



CHAPTER 3

Design Point Analysis

In this Chapter the design point parameters at the supersonic point are selected for a turbojet engine and for the three variable cycle engines selected previously. At the subsonic point the turbines will be matched first and then a performance investigation will be carried out for different bypass ratios, overall pressure ratios and turbine entry temperatures. The effect of engine parameters on the noise level at take-off will be considered.

CHAPTER 3

Design Point Analysis

3.1 Introduction

The selection of the supersonic design point parameters is the first step in designing the variable cycle engine. These parameters will be selected in such a way that they will give the best combination between the fuel consumption and the specific thrust. The choice of these parameters will determine the size of the engine including the turbine non-dimensional mass flows and the nozzle maximum exit area. In the beginning, the choice of these parameters is mainly based on fuel consumption and specific thrust, but lateron this choice could be changed in order to obtain a better matching with the subsonic or take-off design points. This may lead to a better overall engine performance.

At the subsonic point, the turbines are first matched to the supersonic ones. Then the performance investigation are carried out for different bypass ratios, overall pressure ratio and turbine entry temperatures. The engine parameters are selected in order to give the best fuel consumption at this point. The matching with the supersonic point may play a role in changing some of these parameters. Furthermore, the take-off exit jet velocity has an important role in determining the bypass ratio at the subsonic mode.

The noise level at take-off is considered to have an acceptable value when the nozzle exit jet velocity is around 400 m/sec. This value will determine the minimum bypass ratio which will lead to an acceptable noise level at take-off. The use of ejector is not considered in this case.

3.2 Supersonic Design Point Performance

The supersonic design point selection is mainly based on the selection of the best combination of bypass ratio, overall pressure ratio and turbine entry temperature which gives the best trade-off between SFC and the specific thrust. The supersonic performance is carried out at a Mach number of 2.7 and an altitude of 18,750 m, each engine is required to deliver a net thrust of 82,500 N.

Because a turbojet engine is the best candidate for supersonic cruise, a performance study of this engine is carried out in order to compare it with the other variable cycle engines chosen in this work.

Figure 3.1 shows the uninstalled fish-hook curves for the turbojet engine. The effect of turbine entry temperature and overall pressure ratio on SFC and specific thrust are shown in this Figure. The maximum overall pressure ratio is limited by the compressor exit temperature (around 950 K).

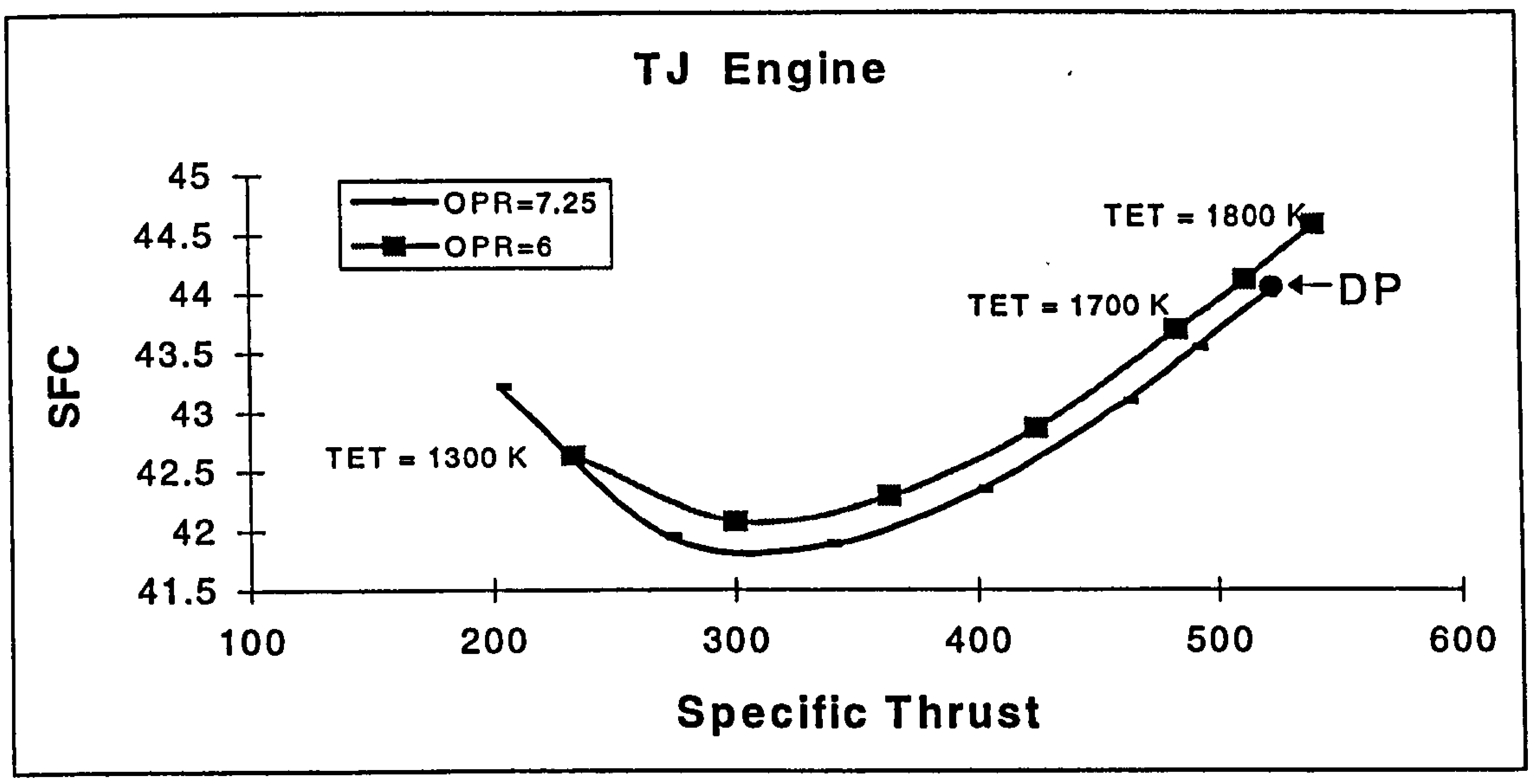


Figure 3.1 Turbojet Engine Supersonic Design Point Performance.

The SFC reach its minimum value when the turbine entry temperature is some where between 1400 K and 1500 K. When this temperature increase from 1500 K to 1800 K the increase in SFC is 5%, and 54% for the specific thrust for the overall pressure ratio of 7.25. The increase in SFC is 6%, and 80% for the specific thrust for the overall pressure ratio of 6 when the turbine entry temperature increase from 1400 K to 1800 K. The increase in specific thrust is much higher than the increase in SFC when the turbine entry temperature increases.

The effect of overall pressure ratio on SFC is an increase of less than 1.4% when the overall pressure ratio decreases by around 18%. The same reduction in overall pressure ratio will increase the specific thrust by 4% when TET=1800 K, and an increase of around 15% when TET=1300 K. The effect of overall pressure ratio on the specific thrust is more apparent when the turbine entry temperature is low, and its effect on SFC is almost independent of this temperature.

Figure 3.2 shows the fish-hook curves for the turbofan-turbojet engine. The effect of bypass ratio and turbine entry temperature on SFC and specific thrust are presented in this Figure. The overall pressure ratio has its maximum value which is limited by the HP compressor exit temperature (OPR=7.05).

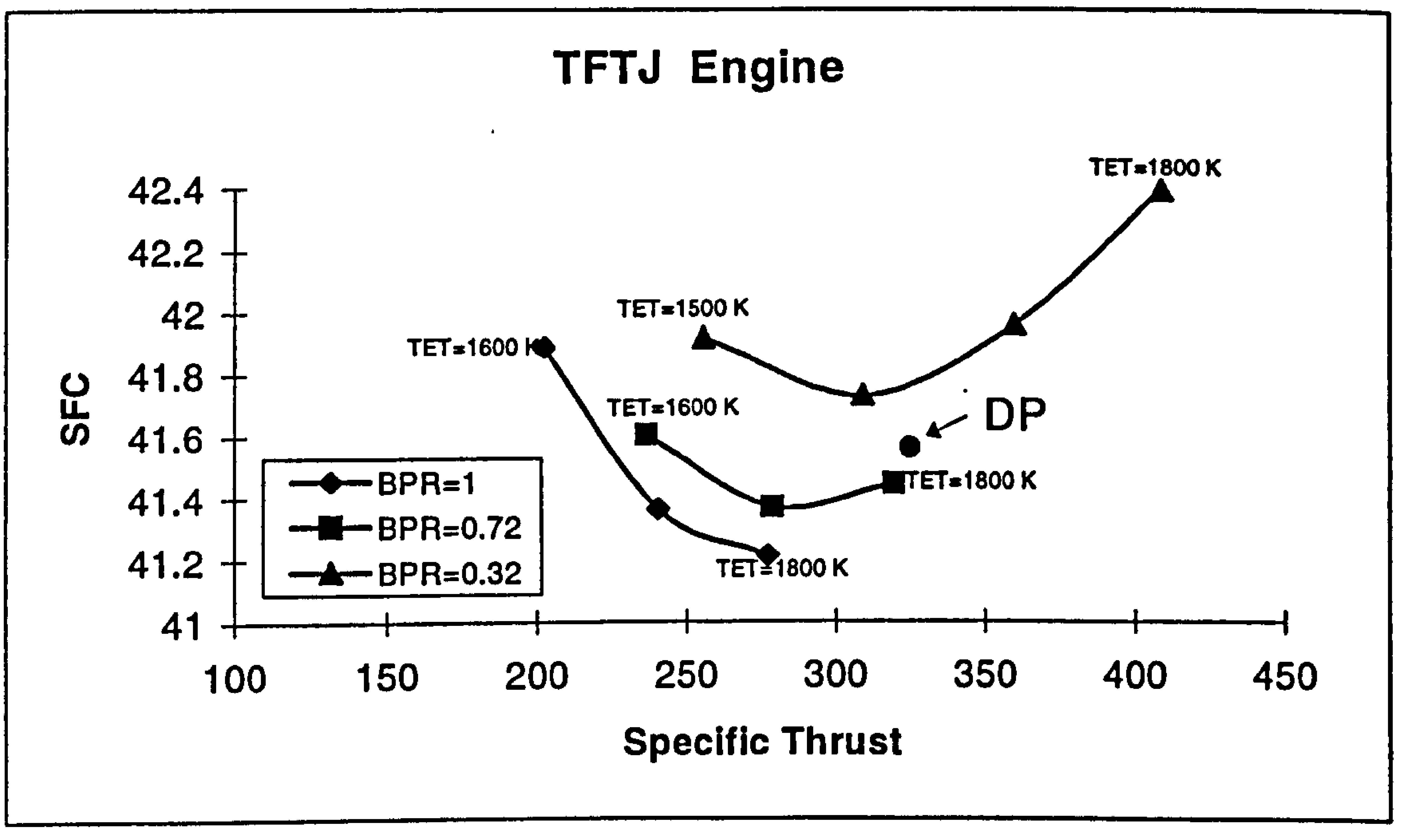


Figure 3.2 Turbofan-Turbojet Engine Supersonic Design Point Performance.

When the bypass ratio increases, the SFC and the specific thrust decrease. When the bypass ratio increases from 0.32 to 1 the SFC and the specific thrust decrease by 3% and 32% respectively for a turbine entry temperature of 1800 K. For a given bypass ratio, the effect of turbine entry temperature on specific thrust is more important than on SFC.

When the bypass ratio increases the turbine entry temperature which gives the minimum SFC increases as well. For the bypass ratio of 0.32, the SFC is minimum when this temperature is 1600 K, they are 1700 K and 1800 K for the bypass ratio 0.72 and 1 respectively.

Figure 3.3 and 3.4 show the fish-hook curves for the Mid-Tandem Fan and the Double bypass engines respectively. The maximum overall pressure ratio is again limited by the HP compressor exit temperature and it has the same vaule for the three engines. These curves have the same patterns as the Turbofan-Turbojet engine.

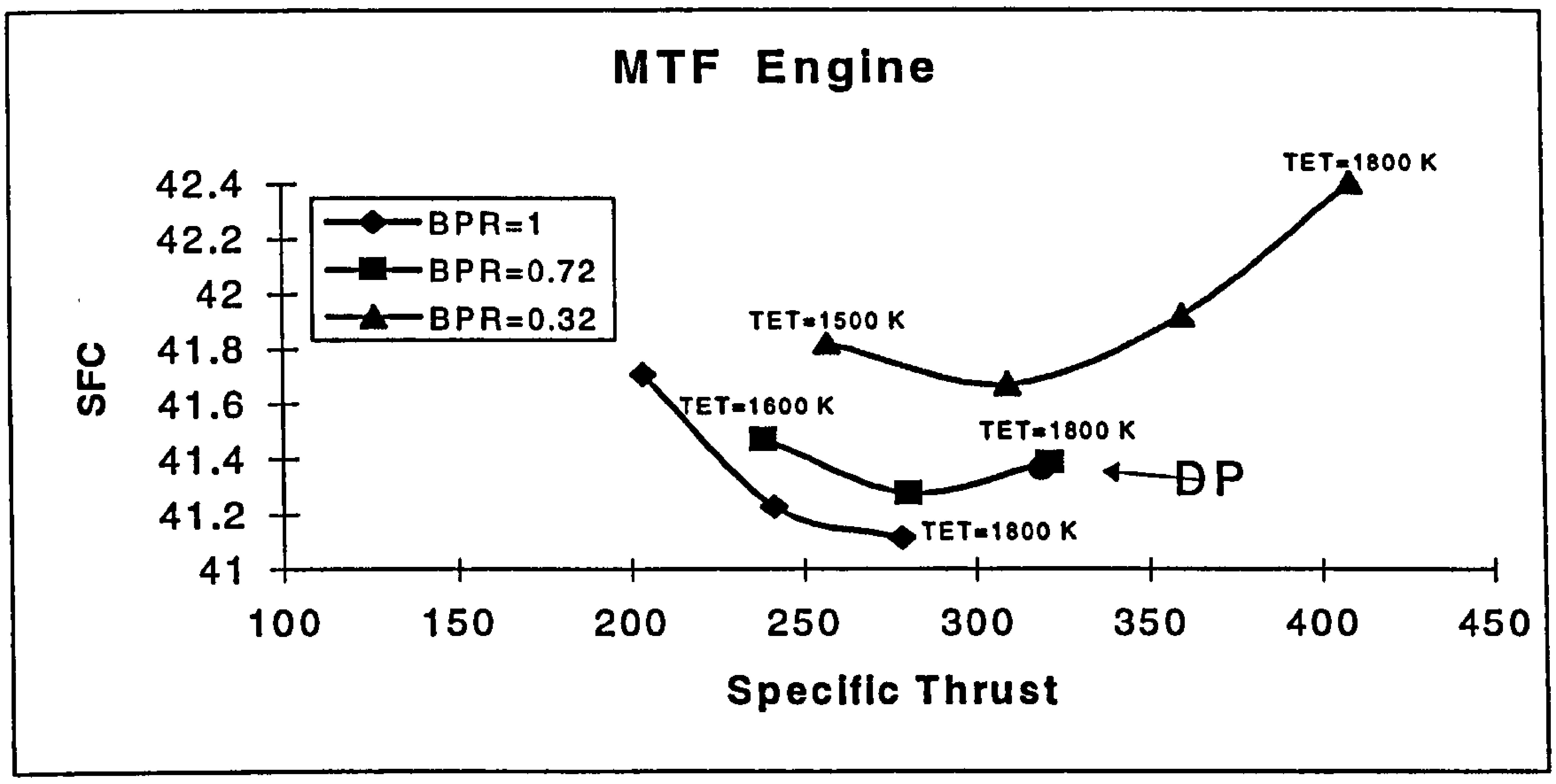


Figure 3.3 Mid-Tandem Fan Supersonic Design Point Performance

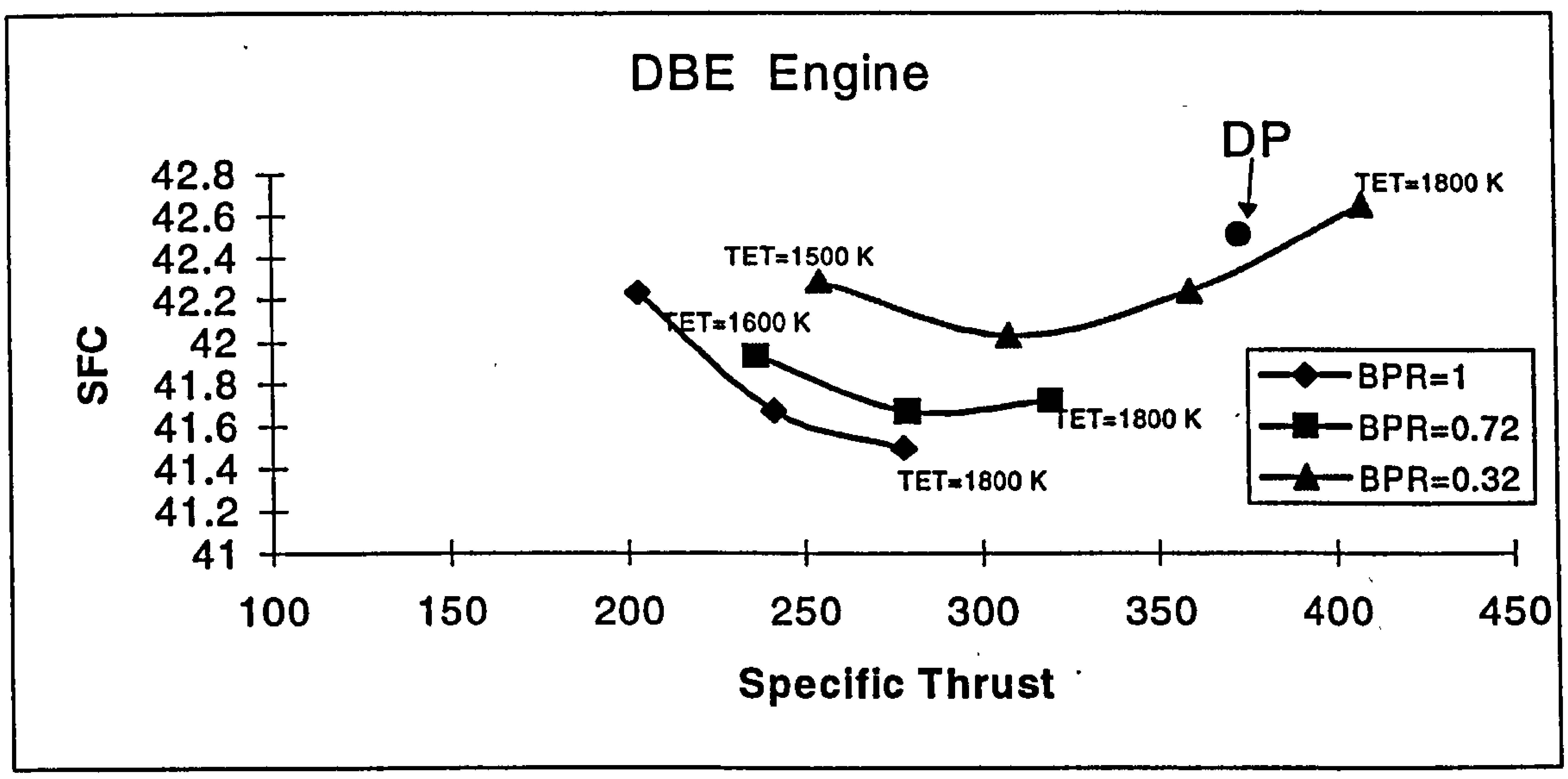


Figure 3.4 Double Bypass Engine Supersonic Design point Performanc.

The design point SFCs and specific thrust are quite similar for the three engines. This result indicates that the different engine component layouts have small effect on the uninstalled supersonic design point performance. They will have a direct effect on the nacelle drag and the engine off-design performance as it will be disscused in Chapter 4.

3.2.1 Supersonic Design Point Selection

The selection of the supersonic design point parameters of a variable cycle engine is based on the following three main criterion:

1- Low fuel consumption and high specific thrust (economic and small).
2- Better matching with the subsonic point engine parameters.
3- Able to meet the noise regulation at take-off.

The supersonic design point parameters are the overall pressure ratio, turbine entry temperature and the bypass ratio. The overall pressure ratio is selected in such a way that the HP compressor exit temperature is around 950 K. The maximum turbine entry temperature of 1800 K is chosen because it gives the highest specific thrust (small engine) and the best thermal efficiency. The selection of these two parameters is mainly controlled by the first criteria.

The selection of the bypass ratio is controlled by the second and third criteria. The choice of the bypass ratio will have a great impact on the size of the engine, especially on the LP compressor frontal area. For the maximum overall pressure ratio and turbine entry temperature, Table 3.1 examines the effect of the bypass ratio on the LP compressor inlet diameter and the maximum mass flow at take-off for the turbofan-turbojet engine. These calculations were done assuming that the LP compressor inlet Mach number is 0.6 and the hub/tip ratio is 0.4.

Engine/bypass ratio	0	0.32	0.72	1
	Diameter (m) / maximum mass flow at take-off (kg/s)			
Turbojet	1.11 / 166			
TFTJ		1.25 / 210	1.42 / 270	1.52 / 310

Table 3.1 Effect Of Bypass Ratio On LP compressor inlet diameter and maximum mass flow at take-off.

Table 3.1 shows that the turbojet engine is the smallest, but the maximum mass flow able to pass throw the engine at take-off is only 166 kg/s where around 450 kg/s is required in order to meet the noise regulation at this point. When the bypass ratio increases from 0.32 to 1 the LP compressor inlet area increases by 48% (22% increase in diameter) and the mass flow at take-off will be increased by the same amount.

The above results can be applied to the LP compressor of the double bypass engine because both engines have the same inlet structural design. For the mid-tandem fan engine, the inlet flow is split between the LP compressor and the fan, the inlet frontal area depends mainly on the design of these two components.

It can be seen from Table 3.1 that even for a bypass ratio of 1 the engine will still not be able to pass the required mass flow at take-off for which the noise level is acceptable. On one hand, increasing the bypass ratio at supersonic point will increase the drag and the weight of the engine and the overall performance will be reduced. On the other hand, the matching with the subsonic point and the noise at take-off will improve. This presents one of the biggest dilemma faced by the variable cycle engine for supersonic civil transport.

A bypass ratio of 0.72 is selected as a compromise between a very low bypass ratio and a medium bypass ratio for the TFTJ and MTF engines. For the DBE a bypass ratio of around 0.45 is selected which is limited by the matching of the LP turbine NDMF. In fact, the DBE has its bypass stream split after the IP compressor and then mixed with the core stream after the LP turbine exit, both streams should have approximately the same mixing total pressure. The LP turbine exit total pressure is a function of the flight conditions and the engine parameters (i.e. TET, OPR and BPR). Table 3.2 shows the variation of the LP turbine exit total pressure in function of the bypass ratio for a Mach 2.7 and TET of 1800 K and the maximum overall pressure ratio.

BPR	**1**	**0.72**	**0.41**	**0.1**
LPC PR	1.35	1.45	1.53	1.66
IPC PR	1.2	1.2	1.2	1.2
HPC PR	4.4	4.01	3.88	3.58
LPT exit Total pressure	2.03	2.15	2.33	2.56

Table 3.2 DBE LPT Exit Total Pressure In Function Of Bypass Ratio

Increasing the bypass ratio will reduce the LP turbine exit total pressure because the mass flow going through the LP and IP compressors is increased and the LP turbine work needed to drive these compressors is increased, for the same TET and OPR. Therefore, increasing the bypass ratio for the DBE is accompanied by a reduction of the bypass stream pressure ratio in order to match the LP turbine exit total pressure, at the same time the HP compressor pressure ratio is increased in order to maintain the same OPR. Consequently, the LP turbine NDMF will increase rapidly and some difficulties were found to match this cycle to the subsonic one. For this reason, the DBE bypass ratio is limited to 0.45 and not 0.72. In fact, the DBE was designed originally for fighter aircrafts where the bypass ratio and Mach number are much less than for supersonic civil transport.

The final selection of the supersonic design point parametrs, for the three engines, is shown in Table 3.3. This selection may not be the best in terms of fuel consumption and specific thrust but it gives a beter overall performance in termas of cycles matching. These parameters will define the turbine non-dimensional mass flows which are important for the matching of the subsonic point.

	Turbojet	TFTJ	MTF	DBE
TET (K)	1800	1800	1800	1800
BPR	0	0.72	0.72	0.46
LPC PR	-	1.67	1.74	1.78
IPC PR	-	1.82	1.8/1.76	1.17
HPC PR	7.05	2.17	2.25	3.18
OPR	7.05	6.6	7.05	6.63
HPT NDMF	760	724	691	735
LPT NDMF	-	1209	1180	1512
Mass Flow (kg/s)	158	254	150/108	221
SFC(mg/N sec)	44.05	41.56	41.36	42.51
Specific Thrust (N/kg/sec)	523	325	319	373
Diameter. m	1.26	1.56	-	1.46

Table 3.3 Supersonic Design Point Parameters Selection

It is worth mentioning that these results are obtained for a fully expanded convergent divergent nozzle. The corresponding nozzle exit area, for the TFTJ, is around 3.75 square meter. It is evident that this area is larger than the inlet compressor face area, therefore, the nozzle exit area has to be reduced and the nozzle is under expaneded. Table 3.4 shows the effect of under expansion on the turbofan-turbojet engine performance. The nozzle coefficient has its maximum value for all the cases. The same level of changes can be applied to the other engines.

	Full Expansion	Under Expansion	Under Expansion
Exit Area (m^2)	3.75	3.05 (19% reduction)	2.74 (28% reduction)
Net Thrust (N)	82500	81568 (1% reduction)	80462(2.5% reduction)
SFC (mg/N sec)	41.56	41.92 (1% increase)	42.50 (2.5% increase)

Table 3.4 Effect Of Nozzle Under Expansion On Engine Performance

3.3 Subsonic Design Point Performance

The first step in the subsonic design point performance is to match the HP turbine with the supersonic mode. Once the HP turbine is matched, the LP turbine non-dimensional mass flow is defined by selecting the HP compressor pressure ratio at this mode. The subsonic performance is carried out at a Mach number of 0.95 and an altitude of 9,150 m, each engine is required to deliver a net thrust of 56,750 N.

Because a turbojet engine is the best candidate for supersonic cruise, the off-design study performance of this engine is carried out in order to compare it with the performance of the other variable cycle engines at this mode. Table 3.5 shows the off-design performance of the turbojet engine selected previously, only TET and the nozzle exit area (fully expanded) are changed in order to obtain the thrust required at the subsonic mode.

	TET(K)/Massflow	SFC (mg/N sec)	Specific Thrust
Turbojet Engine	1230 / 110 kg/s	30.5	521

Table 3.5 Turbojet Off-Design Performance At The Subsonic Mode.

Figure 3.5 shows the turbofan-turbojet engine subsonic design point performance for different bypass ratios, overall pressure ratios and turbine entry temperatures. These parameters are dependent on each other because the HP turbine is matched to the supersonic mode. For a given bypass ratio and turbine entry temperature the overall pressure ratio is calculated in such a way that the HP turbine non-dimensional mass flow remains the same as at the supersonic mode.

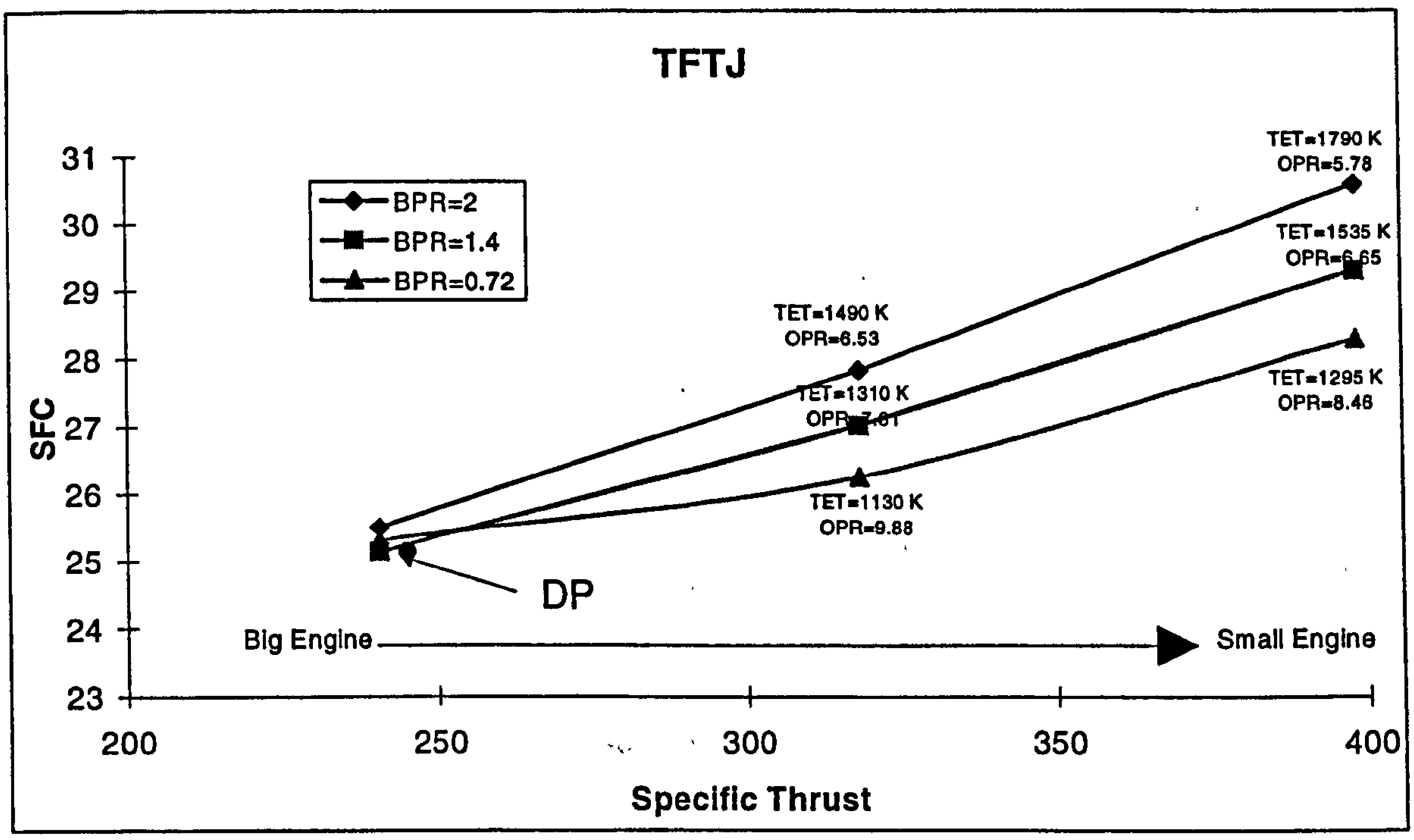


Figure 3.5 Turbofan-Turbojet Engine Subsonic Design point Performance.

Increasing the bypass ratio is coupled with a reduction of the overall pressure ratio and an increase of the turbine entry temperature in order to maintain the inlet HP turbine non-dimensional mass flow constant. The gain in fuel consumption achieved by the increase of the bypass ratio is quickly lost by the reduction of the overall pressure ratio and the increase of the turbine entry temperature. The result is a lower SFC for a lower bypass ratio turbofan which is contradictory to the conventional performance of a turbofan.

For a low specific thrust (high inlet mass flow), the core mass flow is high and the turbine entry temperature is low, consequently, the overall pressure ratio is relatively high in order to maintain a constant NDMF of the HP turbine, this leads to a lower SFC. In this case, the reduction of SFC due to the increase of bypass ratio is less or more compensated by the reduction of overall pressure ratio and the increase of turbine entry temperature. For this reason, the three different bypass ratio engines give almost the same SFC at low specific thrust. A bypass ratio of 1.4 is selected as the design point for the subsonic mode.

Figure 3.6 shows the same characteristics for the mid-tandem fan engine. The curves have the same patterns as Figure 3.5 because the core design of the two engines is the same. Once again, a bypass ratio of 1.4 is selected as the design point for the subsonic mode.

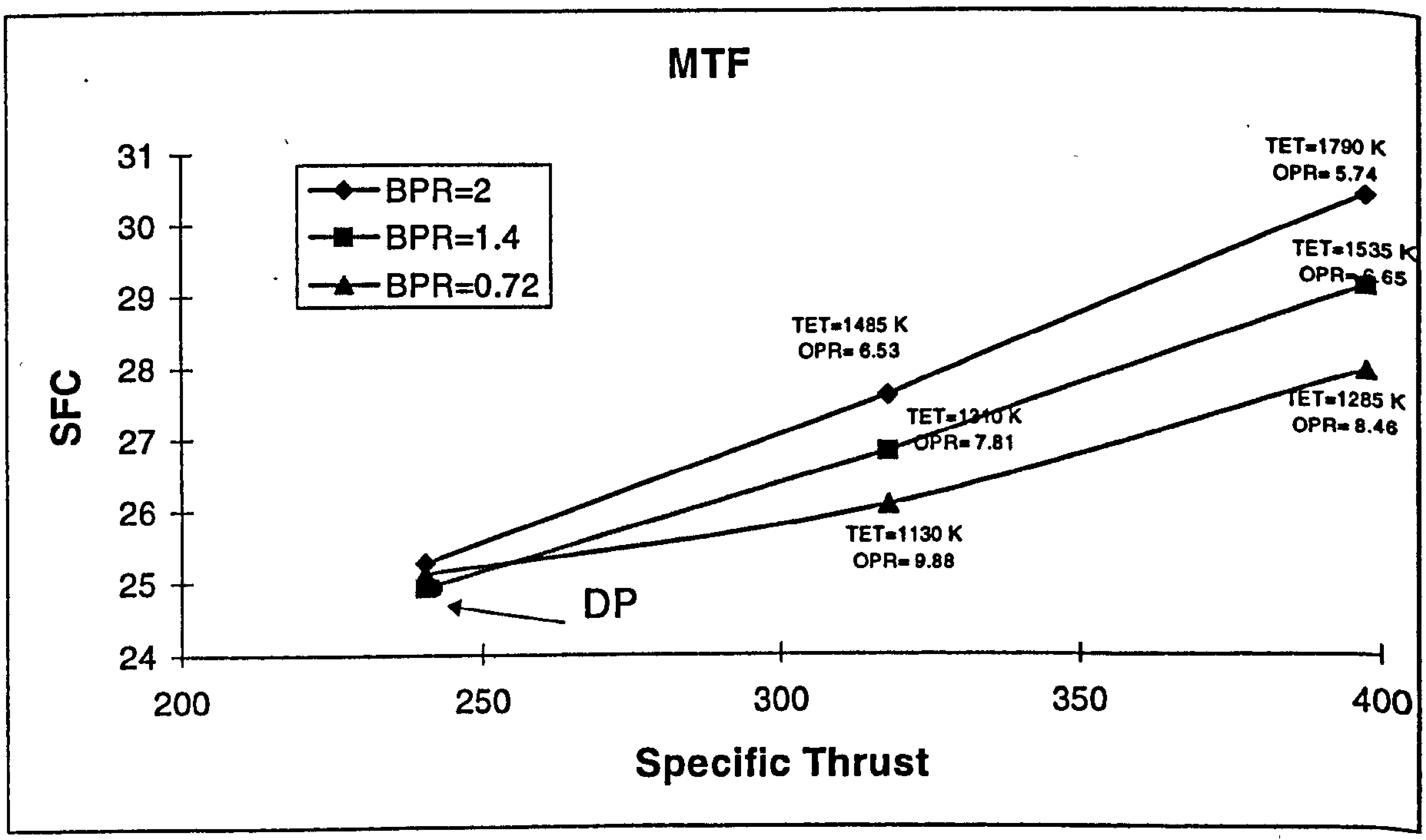


Figure 3.6 Mid-Tandem Fan Engine Subsonic Design point Performance.

For the double bypass engine, the subsonic performance depends directly on the matching of the LP turbine. Because the bypass stream is split after the IP compressor at the supersonic mode, the HP compressor pressure ratio is relatively high in comparison with the two other engines and this leads to a higher LP turbine NDMF at this mode.

The main aim of bleeding the bypass flow after the IP compressor at the supersonic mode is to increase the bypass stream total pressure which is necessary for lower bypass ratio. At subsonic mode, the bypass ratio is higher and the bypass stream is split after the LP compressor in order to reduce the fan pressure ratio. This design is found unnecessary for high flight Mach number where the inlet total pressure is high. This design may be very suitable for lower supersonic Mach number.

Because the double bypass engine LP turbine NDMF at the supersonic mode is high, the range of bypass ratios at subsonic mode where the LP turbine can be matched to the supersonic one is limited to a lower bypass ratio because the core mass flow need to be relatively high.

Table 3.6 shows the variation of SFC in function of bypass ratio for an inlet total mass flow of 250 kg/s (i.e. specific thrust of 229), this value is limited by the matching of the LP turbine, lower inlet total mass flows lead to impossible matching between the cycles. A bypass ratio of 1.27 is selected as the design point for the subsonic mode.

Bypass ratio	**1.52**	**1.3**	**1**
SFC (mg/N sec)	27.69	25.97	25.52
Specific Thrust	229	229	229

Table 3.6 DBE Subsonic Performance.

Table 3.7 shows the final selection of the subsonic and supersonic parameters for all the engines. It can be seen that the SFCs of VCEs are less by around 12% to 18 % in comparison with the turbojet engine at the subsonic mode. At supersonic mode the SFCs are less by around 3.5% to 6%. This represents one of the major achievements of the VCEs. These improvements in SFC should be carefully weighted against the increase in engine size and complexity.

	Supersonic	Take-off
Altitude / Mach number	18750 m / 2.7	0 m / 0
Inlet mass flow required	254 kg/s	454 kg/s
LP compressor Diameter	1.29 m	1.69 m
LP compressor NDMF	0.04443	0.07611

Table 3.8 Conflicting Parameters Between Take-off And Supersonic Points.

The same conclusions can be applied to the DBE because of the similarities with the TFTJ engine. For the MTF, The design of the LP compressor and the fan will define the engine inlet frontal area, this will be seen in Chapter 4.

There are two possibilities to improve the above conflicting requirements. The first is to use an ejector after the nozzle where the nozzle exit velocity can be allowed to reach a higher value and then the total mass flow decreases allowing a better matching with the supersonic point. The use of ejector is outside the scope of this thesis for time considerations. The second possibility is to increase the supersonic design mass flow by increasing the bypass ratio or reducing the TET, both methods will lead to a bigger engine increasing the drag and the weight of the engine.

In this work, the efforts were concentrated on matching a supersonic cycle with a bypass ratio around 0.7 to a subsonic cycle of a bypass ratio around 1.4 without using an ejector at take-off. The engine was oversized in order to meet the noise regulation at take-off. This may not present the best solution but it underlines the major problems for future works.

CHAPTER 4

Variable Geometry Compressor Design And Performance

This Chapter has two main aims, the first is to study the off-design performance of the three VCEs selected previously by using an upgraded version of the Turbomatch program, especially the transition from one mode to another. An overall performance comparison between these engines was undertaken. The second aim is to present a summary of the major aspects on the preliminary aerodynamics design of an axial-flow compressor and turbine and the effects of using a variable stator angle on the compressor maps.

CHAPTER 4

Variable Geometry Compressor Design And Performance

4.1 Introduction

Once the two design points are selected and their turbines matched, Chapter 2, the simulation of VCE will start from take-off passing by the first design point (subsonic mode) and ending at the second design point (supersonic mode). This transition from one mode to the other is done by manipulating the TET, LP and IP compressor variable stators, the nozzle throat and mixing areas. These parameters will be changed in a synchronised manner in such a way that the engine will function in a safe zone of operation on its component maps.

Each one of the three selected VCEs in this work has three compressors, the LP and IP compressors on the LP shaft and the HP compressor on the HP shaft. The LP and IP compressors will play an important role in changing the engine cycle from a low bypass ratio to a medium bypass ratio. They therefore will have some stages with variable stators. The HP compressor and the turbines will have fixed geometries. Variable geometry turbine will be discussed in Chapter 6.

This Chapter has two main aims, the first is to show that a VCE can be simulated using an upgraded version of the Turbomatch program, especially the transition from one mode to another, then applying this to the Turbofan-Turbojet, Mid-Tandem Fan and The Double Bypass engines selected previously. The second aim is to present a summary of the major aspects on the preliminary aerodynamics design of an axial-flow compressor and turbine and the effects of using a variable stator angle on the compressor maps.

The compressor and turbine design and off-design performance are carried out using computer programs available in the SME directory. These programs are:

COMPDES	:	Compressor design point
HOWELL	:	Compressor off-design
TURBGEO	:	Turbine design (mean aerodynamic parameters)
TURBDES	:	Turbine design point (Three-dimensional effect)
TURBOFF	:	Turbine off-design

The component maps generated by the above programs are introduced in the original Turbomatch input files in order to replace the standard maps used previously. This will give a more accurate simulation of the VCE.

A comparison between the uninstalled performance of the three VCEs is carried out, and the fuel bill for two standard missions is estimated as well.

4.2 Variable Cycle Engine Simulation

In previous work it was necessary to use two input files to simulate the two design points, but with the new version of Turbomatch the simulation of a VCE will be achieved using only one input file. The turbine matching and the compressors pressure ratios selection at off-design are done automatically by the program.

It is necessary to start the simulation using the standard compressor and turbine maps that Turbomatch has in its source code. This will be helpful to estimate the amount of variable geometry required by each component before starting the actual design. The maps obtained from the design software could be added to the original input file at a later stage in the simulation process.

4.2.1 Running Turbomatch Program

The simulation of an engine using Turbomatch program needs a starting point, the design point, where the program calculates the main engine characteristics such as the turbine NDMF, nozzle throat and exit areas, mixing areas and the different scaling factors for the standard component maps.

This starting point is usually the supersonic point, where the engine will spend a large part of its operating time. But when simulating VCE, the compressor stators need to be open and closed during off-design simulation. Turbomatch can open the stator by only 10° maximum and closes it by 40° maximum, relative to the design point. At supersonic point the compressor stators are closed. This means they can be open by only 10° at subsonic and take-off points which is insufficient to simulate the VCE through the whole mission profile. Moreover, the engine should be able to pass the required mass flow at take-off in order to meet the noise regulations. That is why the take-off point was chosen to be the starting point for the Turbomatch program where the stators could be closed up to 40°, relative to take-off position, at supersonic point.

This change will require some more information about the engine parameters at take-off point in order to be included in the input file. These parameters are: The total inlet mass flow, TET, and the different compressor pressure ratios. The bypass ratio will be the same as the subsonic point. This is relatively easy to do, bearing in mind that we can run the subsonic design point in off-design, down to take-off point and then the above parameters will be known. Note that the engine will remain matched to the two design points. The relationship between TET and the total inlet mass flow at take-off will be defined in such a way that the nozzle exit velocity, at this point, is about 400 m/sec.

After that, the off-design points can be added to the end of the input file one by one. Every time a point is added, the program should be run in order to obtain the required thrust from the engine and to ensure that all the surge margins are in a safe zone of operational domain. For more details on how to run Turbomatch program see Reference 4.

4.2.2 Results Treatment

The program will produce mainly two output files, the first one will contain detailed information about all the engine components for each off-design point. This file is called FOR002.DAT. The second file will contain the information provided by PLOTBD and PLOTSV bricks and eventually the compressor maps with their different stator angles. This file is called FOR003.DAT and it can be used directly by Microsoft Excel in order to produce charts and the running lines on their compressor maps.

This process will provide a quick way to visualise the position of the running line on the compressor maps and their surge margins. It will also show the shape of the running line during the transition from one mode to another and may indicate that some engine parameters have to be modified at some points in the mission profiles.

4.3 Compressor Design

After cycle calculation, optimisation and analysis, the compressor design and off-design performance is carried out for LP, IP and HP compressors for the three selected VCEs. The first step was to design the compressors as if they had fixed geometry, then the effect of changing the stator angle on the compressor maps was investigated using the Howell program.

4.3.1 Design Point Objectives

As previously mentioned, the design point selected for designing the components is the take-off point, for the reasons mentioned in 4.2.1. In fact, the Howell program showed that the range of closing the stators angle is much higher than the range of opening them. If the supersonic point was selected as the design point, it would be impossible to simulate opening the stator angle by 30° or more using this program. That is the reason for again selecting the take-off point as the design point for designing the compressors. In reality, the stators will have two positions, fully open and fully closed. The design point could be one of these positions or some where between. In our case, the possibility of choosing the supersonic point, fully closed, is not available due to the limitatiom of the Howell program.

Actually, this choice has two implications. The first is that the compressors may be bigger than if they were designed at the supersonic point and that will lead to a higher engine dimensions. The second implication is that the compressor efficiencies at supersonic point may be lower than the target predicted (88%) because the compressors operate at off-design points and that will lead to a higher SFC. One good thing which results from this choice is that the results obtained from this work will have a big margin of safety in terms of size and fuel consumption and the real engine could be smaller and consume less fuel.

The input data for each compressor for the three engines are shown in Table 4.1. This data is drawn from Turbomatch simulations. This data and other complementary data are grouped in an input file which will be used by the Compdes program using the radial equilibrium approach to design the compressor. Although simple radial equilibrium is no longer a major tool for final design of compressors, it can still be used to undertake preliminary design and it is therefore appropriate for this work. In addition, very high accuracy is not important in this work since the aim is to establish that the compressors could in theory, be designed to perform as the cycle demanded. The input and output data files for each compressor for the three engines can be found in appendices B,C and D.

	Pressure Ratio	Target Efficiency	Mass Flow (kg/s)	Inlet Total Pressure (atm)	Inlet Total Temperature (K)
			.TFTJ		
LPC	1.86	88%	453.3	1	288.15
IPC	1.8	88%	188.1	1.86	351.5
HPC	2.54	88%	188.1	3.348	424.15
			MTF		
LPC	1.86	88%	188	1	288.15
FAN	1.86	88%	266	1	288.15
IPC	1.8	88%	188.1	1.86	351.5
HPC	2.54	88%	188.1	3.348	424.15
			DBE		
LPC	1.72	88%	445	1	288.15
IPC	1.25	88%	200.3	1.6	335.21
HPC	4	88%	196.3	2.15	368.6

Table 4.1 Compressors Design Point Parameters.

4.3.2 Design Parameters Analysis

An analysis of aerodynamic parameters was carried out in order to select appropriate values to achieve the target efficiency, using the design programs mentioned above. Appendix E provides ranges in which both aerodynamic and mechanical parameters can be selected without serious problems, as well as the resource to make decisions during the design process. All the parameters selected for designing the different compressors for the three engines can be found in the software input files in the corresponding appendices for each engine.

4.3.3 Compressor Off-design Performance

Having obtained the design point performance of the compressors, the next step was to evaluate their off-design performance and estimate how the actual compressor characteristics would fit within those predicted. A mean radius, stage stacking method to predict the complete performance characteristics of a multistage axial flow compressor, named Howell, was identified in the Turbomachinary Group of the School written by A.R. Howell of NGTE.

Compdes was prepared to provide an output file (PERFORM.INP), which can be selected when Compdes is running, and used as an input file to Howell to avoid lengthy manual procedures. The Howell program will provide two output files, containing the pressure ratio and isentropic efficiency against the non-dimensional mass flow ($m*\sqrt{T}/P$) for ten different non-dimensional speeds ($N/\sqrt{T}$). These files can be used directly by Microsoft Excel to plot the compressor maps. The full Howell output data files are not detailed here, because of their size. Some input data files for the Howell program can be found in Appendices B,C and D.

4.3.4 Variable IGV And Stators Concepts

The main aim of using variable IGV and stator angle is to help the transition from one mode to another. In other words, compressor variable geometry is used here to help changing the bypass ratio from low to medium as the flight speed changes from supersonic to subsonic one. Other benefits could be gained from compressor variable geometry such as improving the performance at off-design, surge margin control and increasing the overall compressor range of operation.

In previous work (Ref. 10), the two modes of operation produced two running lines in different positions on the compressor characteristics. Hence the use of variable stators was strongly indicated for both LP and IP compressors. This not only to improve surge margin, but to reduce mass flows when the engine shifts to the alternative operating mode.

The Howell program was employed to obtain the compressor map at each restagger value, due to its short running time and simple input data file requirement. It calculates overall compressor performance for various air mass flows and compressor speed relationships. Individual stage performance is also calculated. During the investigations of the adjustable stators, the deviation angle of each blade row was assumed independent of stagger angle.

The maximum deviation angle is 30^0, relative to the design point, when closing the stators and 10^o when opening them. The maximum deviation angle was needed when closing the LP compressor stators, the IP compressors needed less deviation angles than the LP compressors. In almost all the cases, two or more stages of variable geometry in each compressor were necessary to correctly position the running lines on their compressor maps. The Howell input files can be found in Appendices B, C and D where the variables for stator deviations are shown.

The above process is a time consuming one. Since the variable geometry simulation had to start each time by using only one stage and the minimum deviation angle and then see if the running lines had a good position on their compressor maps. If not, we had to increase the deviation angle, and if that still does not work, the number of stages, using variable geometry, is increased. Some times we had to modify the initial compressor design in order to obtain a good surge margin for both subsonic and supersonic running lines. For future research it will be important to evaluate the effect of RPM, Hub to Tip ratio and the inlet conditions on the feasibility of variable geometry compressors. The LP compressor is the most critical one, the IP compressor needs less deviation angles and number of variable stages in order to achieve the cycle matching required from it. The HP compressor presents no particular problems.

4.3.5 Results And Discusion

Table 4.2 shows the compressor design parameters obtained from using the Compdes program for each compressor for the three selected engines. At the supersonic mode, the LPC efficiencies for TFTJ and DBE are 81% and 68% respectively and an efficiency of 62% for the Fan of the MTF. They are far below the design target efficiency (88%). The reason for that is the choice of RPM and hub/tip ratio was made in order to obtain a large surge margin at design point in such a way that at off-design, especially at supersonic point, this surge margin will be enough, and an efficiency penalty has to be paid. Another reason to explain this loss of efficiency is due to the fact that at supersonic point the rotational speed ($N/\sqrt{T}$) is low because of the high inlet temperature due to the high flight Mach number. Therefore, the position of the supersonic point on the LP compressor map is relatively far away from the design point.

	LPC	IPC	HPC
Number Of Stages	2	3	3
Number Of Variable Stages (At Least)	2	2	0
Maximum Turning Angle (degrees)	30	10	0
Hub/Tip Ratio	0.4	0.77	0.72
Inlet Tip Diameter (m)	1.84	1.44	1.01
Length (m)	0.64	0.28	0.15
	Sup / Sub / Take-off	Sup / Sub / Take-off	Sup / Sub / Take-off
Pressure Ratio	1.67/ 1.83 /1.86	1.83/ 1.79/ 1.8	2.18/ 2.53 /2.54
Isentropic Efficiency	81/ 86/ 86	87/ 89/ 89	89/ 88/ 88

Table 4.2a TFTJ Compressor Design Parameters

	LPC	FAN	IPC	HPC
Number Of Stages	2	2	4	3
Number Of Variable Stages (At Least)	2	2	2	
Maximum Turnning Angle (degrees)	30	30	10	0
Hub/Tip Ratio	0.5	0.54	0.35	0.7
Inlet Tip Diameter (m)	1.3	1.6	0.98	0.98
Length (m)	0.39	0.33	0.8	0.34
	Sup / Sub / Take-off	Sup / Sub / Take-off	Sup / Sub / Take-off	Sup / Sub / Take-off
Pressure Ratio	1.74/ 1.84/ 1.86	1.76/ 1.85/ 1.83	1.8/ 1.77/ 1.8	2.25/ 2.51/ 2.54
Isentropic Efficiency	88/ 88/ 89	62/ 94/ 85	88/ 90/ 90	89/ 88/ 88

Table 4.2b MTF Compressor Design Parameters

	LPC	IPC	HPC
Number Of Stages	2	1	5
Number Of Variable Stages (At Least)	2	1	0
Maximum Turnning Angle (degrees)	30	10	0
Hub/Tip Ratio	0.35	0.7	0.55
Inlet Tip Diameter	1.82	1.4	1.02
Length	0.68	0.13	0.74
	Sup / Sub / Take-off	Sup / Sub / Take-off	Sup / Sub / Take-off
Pressure Ratio	1.78/ 1.67/ 1.7	1.17/ 1.26/ 1.26	3.18/ 3.96/ 4.0
Isentropic Efficiency	68/ 89/ 89	84/ 88/ 88	88/ 88/ 88

Table 4.2c DBE Compressor Design Parameters

To assess the real effect of variable geometry on component efficiency Compdes and Howell programs should be modified in order to increase the range of opening the stator angle from 10 to around 30 degrees, so the LP compressor can be designed at the supersonic point and then the stators can be open at subsonic and take-off points. The present results show that the use of extensive variable geometry in the LPC is accompanied by a large amount of efficiency losses which is not acceptable. For the other compressors the design efficiency meets almost the target one.

The feasibility of variable geometry compressors depends mainly on the supersonic point position on the LPC map. Reducing the supersonic flight Mach number will increase the chance of using variable geometry without large losses in component efficiencies. Experimental work will be needed to prove the theoretical results.

After having obtained the variable compressor maps for the LP and IP compressors and the fixed map for the HP compressor, it is necessary now to insert them in the initial Turbomatch input file. This process includes regrouping the different Howell output files for each compressor into one file which is inserted into the Turbomatch input file for the corresponding compressor. This will be repeated for each compressor in the engine. A small FORTRAN program was created in order to accelerate this operation. A copy of this program can be found in Appendix A.

4.4 Turbine Design And Performance

The design and off-design performance of the HP and LP turbines have been executed by three computer programs: Turbgeo, Turbdes and Turboff. The design and off-design performance have been assessed on the basis of a mean-diameter treatment.

Because the input parameters for the HP and LP turbines are very similar for the three engines, it was decided to consider only the design and off-design performance for the Turbofan-Turbojet engine's turbines. The main aim of this design is to evaluate the real performance and dimensions of these turbines. The standard Turbomatch turbine maps will be used and not the real maps, for this reason the dimensions of the Turbofan-Turbojet engine turbines will be used later for the other engines sizing in this work.

4.4.1 Design Point Objectives

The design objectives are given by the cycle specifications. These usually include the turbine entry temperature and pressure, turbine outlet pressure, turbine power required (or gas mass flow), the shaft speed, and the fuel air ratio. To achieve these requirements, degrees of freedom are number of stages, work split, annulus area, annulus shape (constant tip or mean or hub diameter), stage reaction, type of vortex flow, radial distribution of work and flow, NGV and rotor/chord ratio and aspect ratio and NGV and rotor profile design. Further degrees of freedom are constrained for reasons of mechanical integrity such as: blade centrifugal stress, maximum diameter for installation and NGV and blade cooling. Table 4.3 gives the turbine inlet parameters, taken from Turbomatch output file, for both HP and LP turbines.

Parameters	HP Turbine	LP Turbine
Inlet Total Pressure (atm)	8.078	4.7
Inlet Total Temperature (K)	1235	1021
Inlet Mass Flow (kg/s)	180	191
Fuel-Air Ratio	0.01697	0.01595
Pressure Ratio	1.72	2.63
Temperature Drop (K)	188	184
Isentropic Efficiency	90%	90%

Table 4.3 Turbine Design Parameters Objectives

To carry out the design, two computer programs were used. The first called Turbgeo (Ref. 10), a pitch line method, provides the rapid determination of design velocity diagrams for turbine stages. It also facilitates the selection of the number of turbine stages and the selection of optimum work split among the stages for any given application. In addition, Turbgeo calculates annulus geometry as well as all mean aerodynamic parameters for preliminary assessment. Moreover, output data for the second program called Turbdes is given directly.

The second called Turbdes, is a free vortex design program approach. This allows for three-dimensional effects. It also estimates design point efficiency and undertakes a simple blade stress analysis.

4.4.2 Design Parameters Analysis

A- Blade Loading Coefficient Ψ

Blade loading coefficient is defined as : $\Psi = \Delta H / U^2 = C_p \Delta T / U^2$

Where ΔH = enthalpy drop, C_p = mean specific heat at constant pressure, ΔT = total temperature drop and U = blade speed.

In general, turbines are designed to achieve efficiency over 90%. Turbine efficiency depends primarily on the shape of the velocity triangles. Smith's efficiency correlation (Figure 4.1) shows that high efficiency is obtained when stage loading and to a lesser extent flow coefficients are low. All of the test turbine efficiency values shown in this chart are corrected to exclude tip clearance losses. Figure 4.1 indicates that the most efficient turbines are designed for ψ between 1 and 2.5. If a low value of ψ is used then more stages are required for a given overall turbine output. For an industrial gas turbine, a low value of ψ is adopted since size and weight are not so important but low SFC is vital. Jet engines, however, must be of low weight and small frontal area. Accordingly, higher values of ψ are adopted.

B- Flow Coefficient ϕ

The flow coefficient is defined as a ratio V_a/U where V_a is the axial velocity and U is the blade speed. Figure 4.2 also indicates that the range of values of ϕ for best turbine efficiency lies between 0.6 and 0.8. For industrial and aircraft gas turbines, ϕ follows the same trends as ψ stated above. The required mean blade speed U_m may be calculated by the following equation knowing the enthalpy drop and blade loading coefficient:

$$U_m = \sqrt{(\Delta H/\psi)}$$

Where $$\Delta H = C_p \Delta T$$

The definition and selectation range of other turbine parameters can be found in Appendix E. All the design parameters selected for HP and LP turbines can be found in the input files in Appendix B.

4.4.3 Off-Design Performance

The off-design performance prediction of the HP and LP turbines have been carried out using the Turboff program (Ref 10). The characteristics of the HP and LP turbines are shown in Figure 4.3 and Figure 4.4 respectively. This shows non-dimensional mass flow ($m\sqrt{T}/P$) and efficiency versus pressure ratio for various values of design non-dimensional speed ($N/\sqrt{T}$).

4.4.4 Results And Discussion

Table 4.4 shows the turbine design parameters obtained using the Turbgeo and Turbdes programs. The turbine input and output files can be found in Appendix B. The HP turbine pressure ratio and efficiency are higher than the ones used in Turbomatch program. On the other hand, the LP turbine pressure ratio and efficiency are lower than the ones used in Turbomatch. It is not in the scope of this work to discuss the compatibility between the different softwares used for this design. However, the results are close enough for the purpose of this project.

Parameters	HP Turbine	LP Turbine
Number of Stages	1	2
Inlet Tip Diameter (m)	1.03	1.32
Outlet Tip Diameter (m)	1.48	1.55
Turbine Length (m)	0.15	0.23
	Sup/ Sub/ Take-off	Sup/ Sub/ Take-off
Pressure Ratio	1.68/ 1.72/ 2.08	2.35/ 2.6/ 2.44
Isentropic Efficiency (%)	90/ 90/ 93	91/ 90/ 89

Table 4.4 TFTJ Turbine Design Parameters.

4.5 Results And Performance

The results can be classified in three categories, the overall engine performance including SFC and specific thrust, then the variable geometry component characteristics and finally the compressor maps with their running lines. A comparison between the three engines, in terms of size, is given.

These results show in detail each engine's performance at three specific points, take-off, subsonic and supersonic points. Engine performance at other points of the mission profile has an importance of the second order. Figure 4.16 shows the net thrust and SFC at different off-design Mach numbers for the three engines.

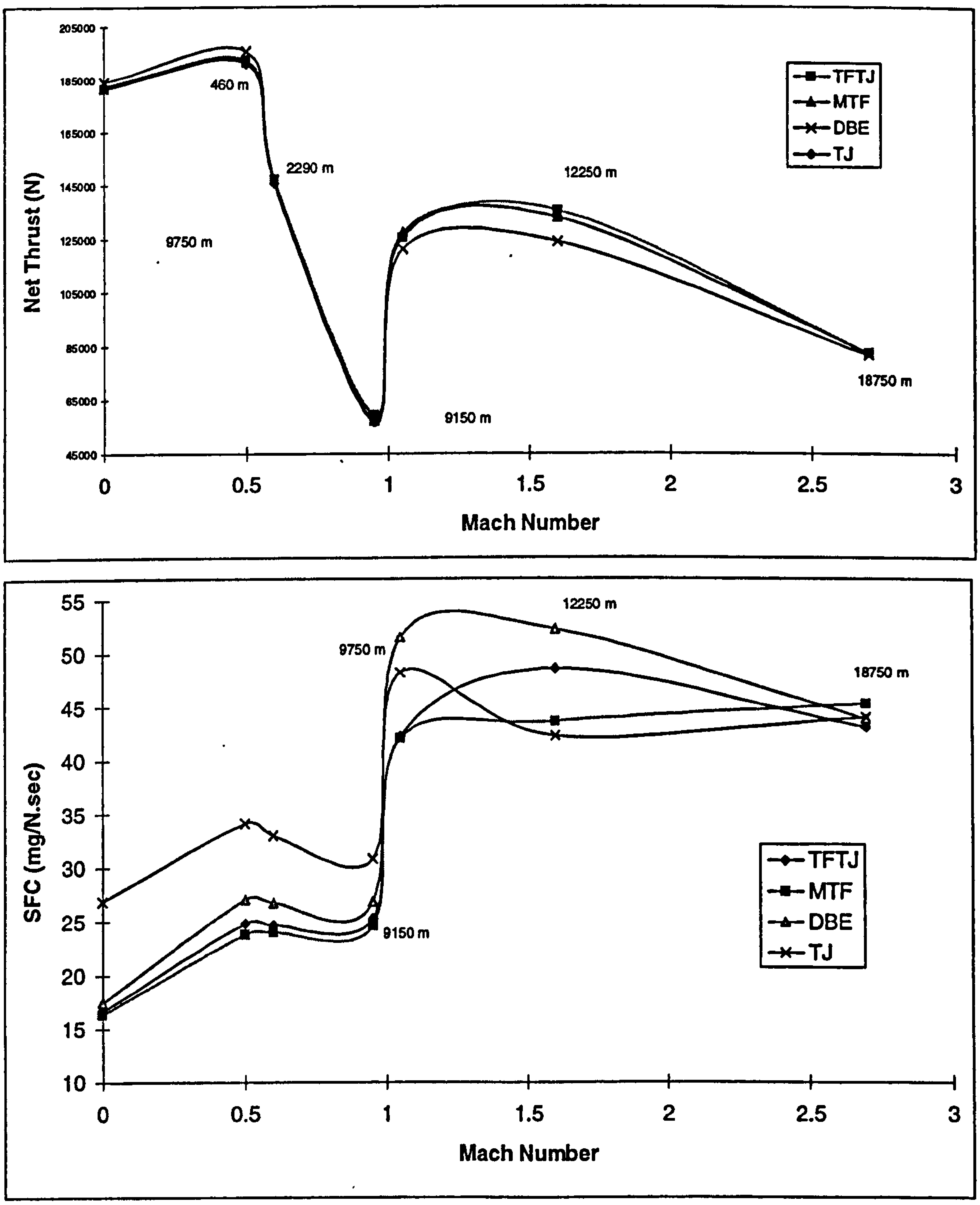


Figure 4.16 The Effect Of Mach Number On SFC And Net Thrust At Off-Design Points.

4.5.1- Performance

Table 4.5 shows the different engine parameters and performance at take-off, subsonic and supersonic points for the three engines. They are the uninstalled engine performance, the installed performance will be discussed in Chapter 5.

ENGINE	TFTJ	MTF	DBE
	Sup./ Sub./ Take-off	Sup./ Sub./ Take-off	Sup./ Sub./ Take-off
Altitude (m)	18750/ 9150/ 0	18750/ 9150/ 0	18750/ 9150/ 0
Mach Number	2.7/ 0.95/ 0	2.7/ 0.95/ 0	2.7/ 0.95/ 0
TET (K)	1800/ 1160/ 1240	1800/ 1150/ 1230	1800/ 1150/ 1240
LPC Pressure Ratio	1.67/ 1.83/ 1.86	1.74/ 1.84/ 1.86	1.73/ 1.68/ 1.7
IPC Pressure Ratio	1.83/ 1.78/ 1.8	1.8/ 1.77/ 1.8	1.17/ 1.26/ 1.26
Fan Pressure Ratio	-	1.76/ 1.85/ 1.83	-
Overall Pressure Ratio	6.65/ 8.3/ 8.5	7/ 8.18/ 8.5	6.4/ 8.4/ 8.6
Bypass Ratio	0.72/ 1.42/ 1.4	0.71/ 1.42/ 1.4	0.46/ 1.27/ 1.27
Thrust (kN)	82.5/ 59/ 182	82.5/ 59/ 182.5	81.8/ 57.4/ 182.5
Nozzle Exit Velocity (m/s)	1078/ 536/ 406	1039/ 542/ 408	1150/ 680/ 550
SFC (mg/N.sec)	43.1/ 25.2/ 16.6	45.3/ 24.6/ 16.3	43.9/ 26.8/ 17.3
Specific Thrust (m/s)	308/ 240/ 401	295/ 246/ 403	349/ 237/ 410

Table 4.5 Performance At Supersonic, Subsonic And Take-off points.

During take-off and climb to subsonic cruise, the compressor stators remain at the same setting. Only the nozzle exit area is increased in order to produce a suitable expansion. During climb and acceleration from subsonic to supersonic speeds, the compressor stators, the mixing areas and the nozzle throat areas are changed to modify the cycle from a relatively high bypass ratio to a low bypass ratio. The nozzle exit area continues to increase until it reaches its maximum value, equal, more or less, to the compressor inlet area. Beyond that point the nozzle is under expanded.

The bypass ratio remains almost the same at take-off and subsonic points. During climb and acceleration it decreases gradually until it reaches its design value at supersonic cruise. The reduction achieved is approximately 100% (from 1.4 to 0.71), further reduction in the bypass ratio would lead to a surge problems in the LP compressor and in the mixing area.

The TETs at take-off and subsonic points are relatively low, (around 1240 K at take-off and 1160 K at subsonic points), this is because the mass flow is kept high in order to keep the nozzle exit velocity low thus reducing the noise level at take-off, and in order to achieve a better intake matching at subsonic point. This will have a good effect on engine life and intake matching, but it will have a bad effect on matching the engine as a whole because it will increase the gap between the running lines, for the LP compressor, at subsonic and supersonic points.

In this transonic phase, the after burner is used to increase the thrust in order to meet the mission requirements. At supersonic cruise however the engine is again in the dry mode. It is pertinent to point out that whenever the afterburner is switched on, its temperature is always lower than 1650 K and most of the time below 1400 K. This makes the control of the emissions of oxides of nitrogen a viable proposition.

These preliminary results indicate that the Turbofan-Turbojet engine has a small advantage in terms of fuel consumption. Its SFC is higher by 2.5% than the MTF at subsonic point, but lower by 5% at the supersonic point. The Mid Tandem Fan appears to have the lowest SFC at subsonic point but the highest at supersonic point. The DBE does not meet the noise regulations at take-off due to its high nozzle exit velocity.

Table 4.6 shows the different design point SFCs and specific thrusts when the components have the target and the actual efficiencies. At supersonic point, the difference is remarkable because of the large losses in component efficiencies, but at subsonic point the results are almost the same due to the similarities in component efficiency.

	TFTJ	MTF	DBE
	Sup.. / Sub.	Sup.. / Sub.	Sup.. / Sub.
	Design Point (target efficiency)		
SFC	41.56 / 25.14	41.36 / 24.94	42.51 / 26.94
Specific Thrust	325 / 240	319 / 242	373 / 236
	Design Point (Actual Efficiency)		
SFC	43.1/ 25.2	45.3/ 24.6	43.9/ 26.8
Specific Thrust	308/ 240	295/ 246	349/ 237

Table 4.6 Effect Of Target and Actual Efficiencies On Engines Performance.

These differences in SFCs do not come from a genuine superiority in the engine designs. They come from the different component efficiencies in design and off-design points.

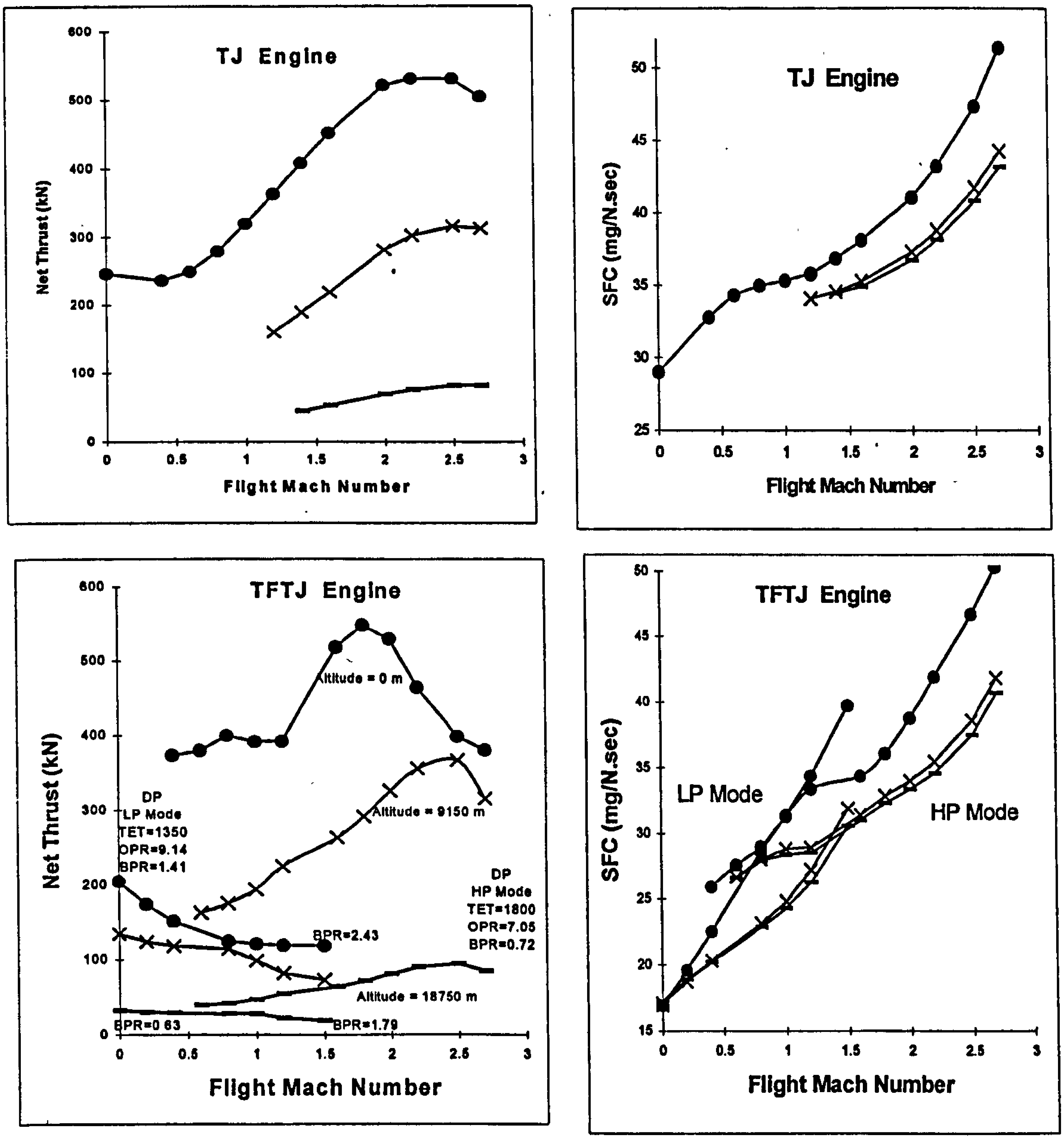


Figure 4.17 Effect Of Altitude and Mach Number On Net Thrust And SFC.

Figure 4.17 shows the variation of net thrust and SFC as a function of altitude and Mach number for the Turbojet and the HP and LP modes of the TFTJ engines. The Design point of the Turbojet and the HP mode of the TFTJ engines is the supersonic one. The take-off point is the design point for the LP mode of the TFTJ engine. The TET and the nozzle throat, at off-design, remains constant for each mode. The LP compressor rotational speed (PCN) and the surge margin are selected in such a way that the non-dimensional rotational speed ($N/\sqrt{T}$) remains within the compressor map limits. The change of these parameters could affect slightly the results.

Increasing the altitude would decrease the net thrust significantly. The inlet total pressure decreases with increasing the altitude, consequently the inlet HP turbine and nozzle total pressure would decrease leading to a decrease in the total mass flow in order to maintain the HPT and nozzle NDMF constant. This decrease in mass flow will reduce the net thrust significantly. This reduction in net thrust is accompanied by a reduction in SFC due to the reduction of core mass flow which needs less fuel flow for the same TET. The reduction in fuel flow is faster than the reduction of net thrust giving lower SFC.

Decreasing the flight Mach number would reduce the net thrust for the Turbojet and the HP mode of the TFTJ engines. Decreasing the Mach number will decrease the inlet total temperature and pressure. The effect of decreasing the Mach number is similar to the effect of increasing the altitude on the HPT behaviour. The reduction in the gross thrust, due to the reduction of total mass flow, is faster than the reduction of the intake momentum drag resulting in a decrease in the net thrust. The reason for the reduction of SFC is the same as above.

For the LP mode of the TFTJ engine the opposite effect happens. Increasing the Mach number would result in a decrease in the net thrust and an increase in the SFC. Increasing the Mach number would increase the inlet total temperature and pressure, the total mass flow will increase in order to maintain the HPT and nozzle NDMF constant, consequently the gross thrust will increase but the intake momentum drag will increase faster than the gross thrust resulting in a decrease in the net thrust.

Increasing the flight Mach number would increase the bypass ratio while increasing the altitude would decrease it. This is due to the fact that the changes in total pressure at the inlet of the HPT and nozzle are not proportionate. Accordingly, the requirement for changes in the core mass flow can be higher or lower than the requirement for changes in the total mass flow. This would lead to a higher or lower bypass ratio when the altitude and Mach number vary.

For the TFTJ engine, the LP mode shows a much lower SFC than the HP mode when the flight mach number is less than 1. On the other hand, the subsonic SFC of the TFTJ engine is much lower than the subsonic SFC of the Turbojet. This again shows the advantages of using VCE for mixed subsonic and supersonic cruises and the reduction in fuel consumption compared to a traditional Turbojet.

4.5.2- Variable Geometry Components

Table 4.7 shows the variable geometry requirements of the three engines. As mentioned earlier, the main aim of the variable geometry components is to change the bypass ratio when the engine shifts from one mode to another. The three parameters used to achieve this aim are: the LP and IP compressor variable stators, nozzle throat and exit area and the mixing area. The nozzle and mixing areas are already known from the design point calculations. The aim of the off-design running is to demonstrate that the engine can make the transition from one mode to other without any major problems.

It can be seen that the LP compressors and fans invariably have a relatively large requirement to vary the stator angle. As far as the IPCs are concerned, they are all similar although the TFTJ has a small advantage.

An examination of the nozzle area change requirements shows that the variable geometry requirement for all engines is approximately similar, with the largest requirement being that of the MTF. As far as the nozzle exit area is concerned, the DBE has the largest requirement, of the order of 300 percent. This is due to the large change in mass flow going through the core engine when the cycle changes from one mode to another.

	TFTJ	MTF	DBE
LPC stator angle change (degrees)	30	10	30
FAN stator angle change (degrees)	n/a	30	n/a
IPC stator angle change (degrees)	5	10	10
HPC stator angle change (degrees)	0	0	0
Nozzle throat area (min) (m^2)	1.07	1.12	0.73
Nozzle throat area (max) (m^2)	1.52	1.85	1.03
Nozzle exit area (min) (m^2)	1.53	1.51	0.73
Nozzle exit area (max) (m^2)	2.7	2.40	3.0
Mixer area Cold (min) (m^2)	0.45	0.36	0.007/0.175
Mixer area Cold (max) (m^2)	1.008	1.007	0.673/0.013

Table 4.7 Variable Geometry Component Parameters

An examination of the mixer area variation reveals that apparently there is a large requirement for variable geometry. However, because the mixer is placed at a large engine diameter the change in flow area could be achieved relatively easily. At the supersonic mode, the mixer tip and hub diameters are expected to be around 1.8 m and

1.63 m respectively. Reducing the hub diameter by 11 % would increase the mixing area by around 100 %.

These parameters have to be manipulated, during the transition mode, in a synchronised manner in such a way that the engine will function safely and smoothly throughout the transition period. These parameter manipulations are gradually introduced in the Turbomatch input file during the off-design calculations. It is important during this phase that component efficiencies, thrust levels and rotational speeds remain at satisfactory levels.

4.5.3- Compressor Maps And Running Lines

The compressor variable geometry maps were obtained directly from the Howell program output files. These maps were introduced in Turbomatch input files and are used throughout the flight envelope. The running lines were obtained from Turbomatch output files. The engine parameters which permit the cycle to change from one mode to other are:

TET or Fuel Flow
LP and IP compressor variable stators
Nozzle throat area
Mixing areas.

In addition to the above, other flight mission parameters are changed during the off-design calculations, they include altitude and Mach number. The use of after burner at some points of the mission profile will not affect the running line positions.

Figures 4.5 and 4.6 show the running lines on the LP and IP compressor maps for the TFTJ engine. It can be noticed that surge margins are acceptable on both maps despite the awkward shape of the running lines. To achieve this a significant movement of the variable stators is required. This also gives satisfactory compressor efficiencies.

Figures 4.8, 4.9 and 4.10 show the running lines on the LP, FAN and IP compressor for the MTF engine. The surge margins on the LP and IP compressor maps are acceptable. It is very low on the FAN. Its efficiency drops to 62 % at supersonic cruise. This is because the bypass ratio has to be reduced in order to increase the specific thrust at this point . The large variation in the operating point causing these effects can be seen in Figure 4.9. This is clearly not acceptable and further investigations in designing the FAN are needed.

Figures 4.12 and 4.13 show the running lines for LP and IP compressors for the DBE. The surge margin in both is acceptable. The LP compressor efficiency drops to 68 % at supersonic cruise while the IP compressor efficiency remains at a satisfactory level. The extraordinary shape of the IP compressor running line is due to difficulties in matching between the DBE cycles while trying to change the bypass ratio. As in the case of the MTF, more attention to detail is required in component design. The very strange path on the IPC map is due to the changes in bypass ratios and nozzle areas. The shape of the running lines during the transition phase is not fixed and it could be modified using the above parameters in order to meet any particular considerations for the transient performance. But the slope of the lines at subsonic and supersonic modes is fixed and defined by the mission profile requirements.

The running lines of the H.P compressors of all engines present no particular problem because the turbines are of fixed geometry. The HP turbine is operating between choked nozzles, therefore the HPC running line is a straight line on the compressor map. Using a variable geometry in the LP turbine may change the shape of this running line and the use of variable geomtry in the HP compressor will be necessary. The different running lines are shown in Figures 4.7, 4.11 and 4.14 for TFTJ, MTD and DBE respectively.

4.5.4- Engine Size

The diameter and length of each compressor and turbine are calculated by the software used in this work. The length of the combustion chamber and the ducts are estimated. Table 4.8 gives the components size for the three engines.

The TFTJ seems to be 10 % shorter than the other two engines, this is due to its simplicity. The lengths of intake and nozzles are not taken into consideration in this comparison, they will be discussed in detail in Chapter 5. The software used here is very powerful but not the most advanced. In practice the components will be shorter and smaller, but the results shown are a useful comparison between the engines.

	TFTJ	MTF	DBE
	diameter/length(m)	diameter/length(m)	diameter/length(m)
LPC	1.84/0.64	1.3/0.39	1.82/0.68
IPC	1.44/0.28	0.98/0.8	1.4/0.125
FAN		1.6/0.33	-
HPC	1.01/0.33	0.98/0.34	1.02/0.74
C.C	0.5	0.5	0.5
HPT	1.03/0.145	1.03/0.145	1.03/0.145
LPT	1.32/0.23	1.32/0.23	1.32/0.23
DUCTS (m)	0.325	0.481	0.484
TOTAL LENGTH (m)	2.55	2.886	2.904

Table 4.8 Diameter And Length Of Different Components

4.6 Fuel Bill

An important performance parameter is the fuel consumption of the aircraft on a typical mission. This will have an important influence on the take-off weight of the vehicle. To compare the three engines under investigation, two journeys were selected to calculate the fuel consumption. The first is between London and Los Angeles, where the first part of the flight is at supersonic speed, over the sea, while the second part is at subsonic speed overland. The second journey is between Los Angeles and Sydney, where the whole flight is at supersonic cruise Figure 4.15.

The calculation of the fuel bill consisted of the evaluation of the amount of fuel consumed at the end of each part of the mission profile. The method is explained in more detail in Reference 3. Table 4.9 shows the final results for the three engines. Normally each mission profile is divided into the following parts.

a- Taxi & Takeoff
b- Climb and acceleration
c- Subsonic Cruise
d- Supersonic Cruise
e- Decent & Deceleration
f- Land & Taxi

The time spent for each part for the two mission profiles is given in Figure 4.15. The SFC can be obtained from Turbomatch off-design calculations, and consequently the fuel flow is known at each point of the mission profile. An average value is taken between the start and the end of each part.

	Fuel Consumed (kg)				
	time(min)	TJ	DBE	MTF	TFTJ
Taxi&Take-off	10	5028.3	4433.5	4323.0	4357.9
Climb&Acceleration	20	25766.1	23683.8	22300.2	22634.3
Supersonic Cruise	110	83996.9	83851.8	87170.7	83407.2
Descent &Deceleration	10	3600.0	3600	3600	3600
Subsonic Cruise	220	62034.3	55717.8	51296.9	53244.1
Descent&Deceleration	20	7200.0	7200	7200	7200
Land & Taxi	10	3600.0	3600	3600	3600
TOTAL	400	191226	182087	179491	178044

Table 4.9a (Mission I) Fuel Bill Comparison

	Fuel Consumed (kg)				
	time(min)	TJ	DBE	MTF	TFTJ
Taxi&Take-off	10	5028.3	4433.5	4323.0	4357.9
Climb&Acceleration	20	25766.1	23683.8	22300.2	22634.3
Supersonic Cruise	230	150647.4	150635	155713	150050
Descent &Deceleration	0		0	0	0
Subsonic Cruise	0		0	0	0
Descent&Deceleration	30	10800.0	10800	10800	10800
Land & Taxi	10	3600.0	3600	3600	3600
TOTAL	280	170076	169469	174436	168808

Table 4.9b (Mission II) Fuel Bill Comparison

The Turbojet engine has the highest fuel bill for mission 1 , where the important part of the flight is carried out at subsonic cruise. It is 5% higher than the worst VCE, this is due to the poor performance of the Turbojet at subsonic flight. For mission 2, the Turbojet performance is very similar to the other VCEs. Furthermore it is much smaller than the others, so the nacelle drag will be reduced, but it is far away from meeting the noise regulation at take-off because of its high nozzle exit velocity.

It can be seen from Table 4.9, resulting from uninstalled SFCs, that the fuel consumption of all the VCE engines is very similar. In the case of the first route, London to Los Angeles, the difference between the best (TFTJ) and the worst (DBE) engine is slightly more than 2 percent. In the other route, Los Angeles to Sydney, the difference is of the same order of magnitude, but slightly larger. In this case, the best engine (TFTJ) is nearly 3 percent better than the worst (MTF) engine.

4.6.1 Component Efficiencies Sensitivity Study

The component efficiencies affect directly the fuel consumption of the engine. Therefore, it was necessary to evaluate the effect of the component deterioration on the fuel bill for the two above missions. The components chosen for this study are the compressors, turbines and the intake. Each component deterioration was simulated separately from the other components. The component design values were reduced by 2% and then the engine is run at off-design with the new deteriorated values.

The first study consisted of reducing all the compressor isentropic efficiencies by approximately 2% relative to the design value, the same thing applied to all the turbine isentropic efficiencies and finally the intake pressure recovery was reduced by 2% relative to the standard value used by the Turbomatch program.

These reductions in component efficiencies led to the reduction in the net thrust obtained from the engine at the different points in the flight envelope. Therefore, it was necessary to increase TET in order to restore the net thrust required at each point and, by consequence, the engine life will be shortened. New SFCs were obtained and the fuel bill was recalculated for the new values. Table 4.10 shows the new fuel bills for the two mission profiles.

The increase in TET in order to restore the net thrust is around 25 to 30 degrees for each component deterioration, this amount could be increased rapidly if all the components were deteriorated together and the life of the engine, especially the HP turbine, will be dramatically reduced. This again shows the importance of component efficiencies in future SST.

The effect of component deterioration on the SFCs at the subsonic cruise is much higher than on the SFCs at the supersonic cruise. This is because the overall pressure ratio at the subsonic cruise (about 8.3) is greater than the overall pressure ratio at the supersonic cruise (about 6.6). Therefore, the component outlet temperatures are more sensitive to efficiency deterioration at subsonic cruise than at supersonic one. Consequently, the fuel bill for the mission London-Los Angeles, which has mixed subsonic and supersonic cruises, is higher than the mission Los Angeles-Sydney which has only supersonic cruise. The average increase in SFCs at supersonic cruise is about 0.4%, but it is about 1.8 % at the subsonic cruise. The MTF showed that its SFC is not very sensitive to the component deterioration at the supersonic cruise, the increase in SFC is less than 0.2%.

	Fuel Bill (kg)					
	London-LosAngeles			LosAngeles-Sydney		
	DBE	MTF	TFTJ	DBE	MTF	TFTJ
Design Values	182087	179491	178044	169469	174436	168808
All compressors deteriorated by 2%/	183145	180683	179273	169986	174566	169200
Difference relative to design	1058	1192	1230	517	131	392
New TETs (K)	1826	1829	1833	1826	1829	1833
All turbines deteriorated by 2%	183176	180669	179500	169797	174563	169562
Difference relative to design	1089	1178	1457	328	128	754
New TETs (K)	1816	1819	1825	1816	1819	1825
Intake pressure recovery deteriorated by 2%	183390	180330	179123	169915	174604	169267
Difference relative to design	1303	840	1079	447	168	459
New TETs (K)	1823	1823	1820	1823	1823	1820

Table 4.10 Effect Of Component Deterioration On Fuel Bill.

4.7 Conclusion

The three VCEs showed large improvements, in comparisson with a pure turbojet engine, in the following points:

1- A significant reduction in SFCs at subsonic cruise(15% to 20% relative to a Turbojet).
2- The jet velocity at take-off was reduced by over 50% relative to the turbojet engine.

However, these improvements are accompanied by an increase in engine size and complexity. For the L.P, I.P and FAN compressors, there is a significant drop in efficiency at the supersonic mode due to the extensive use of variable geometry at this point. However, this problem is due to the fact that the compressors were designed at the take-off point in order to be able to simulate them using Turbomatch and the existing design softwares available in the SME Department. In reality, they would be designed at the supersonic point where the efficiency is maximum, in this case the existing softwares will need to be modified in order to overcome this problem in the future.

The component efficiencies in design and off-design will play an important role in the selection of the future SST powerplant, especially the compressor variable stators where they could vary up to 30° relative to design point for the LP compressor and up to 10° for the IP compressor. The effect of changing the stators angles on the efficiency should be analysed in more detail for the LP and IP compressors.

The control of changing the mixing and the nozzle throat and exit areas, with changing the variable stators, will be fundamental in changing the cycle from low bypass ratio to medium bypass ratio. The area ratio changes from one mode to other could reach 2.8 for the mixing area, 1.6 for the nozzle throat area and 4 or more for the nozzle exit area. The effects of these changes on mechanical integrity should be analysed.

As stated earlier, noise was not analysed explicitly although take-off nozzle exit velocities were, whenever possible, limited to 400 m/s. The noise signature of the aircraft will be critical to its success, so a detailed analysis is required. All performance figures are for the uninstalled engine.

These preliminary results indicate that the three engines are quite similar in terms of general suitability. The Mid Tandem Fan appears to be an attractive proposition from the point of view of sizing, however this comes with a small penalty in fuel consumption. The DBE does not meet the noise regulations at take-off due to its high nozzle exit velocity. Moreover its structural design, bleeding the bypass flow after the IP at supersonic mode and after the LP at subsonic mode, was found not very suitable for a Mach 2.7 and an altitude of 18750 m due to problems in matching the LP turbine NDMF and the mixing of total pressures of the bypass and core streams.

The present results showed that the extensive use of variable geometry in the LP compressor will lead to a significant loss in efficiency. This is due to the large variations in the flight conditions. This problem can be improved by reducing the supersonic Mach number and altitude where the inlet conditions will change in a favourable direction.

FIGURE 4.1 Compressor Stage Loading Chart

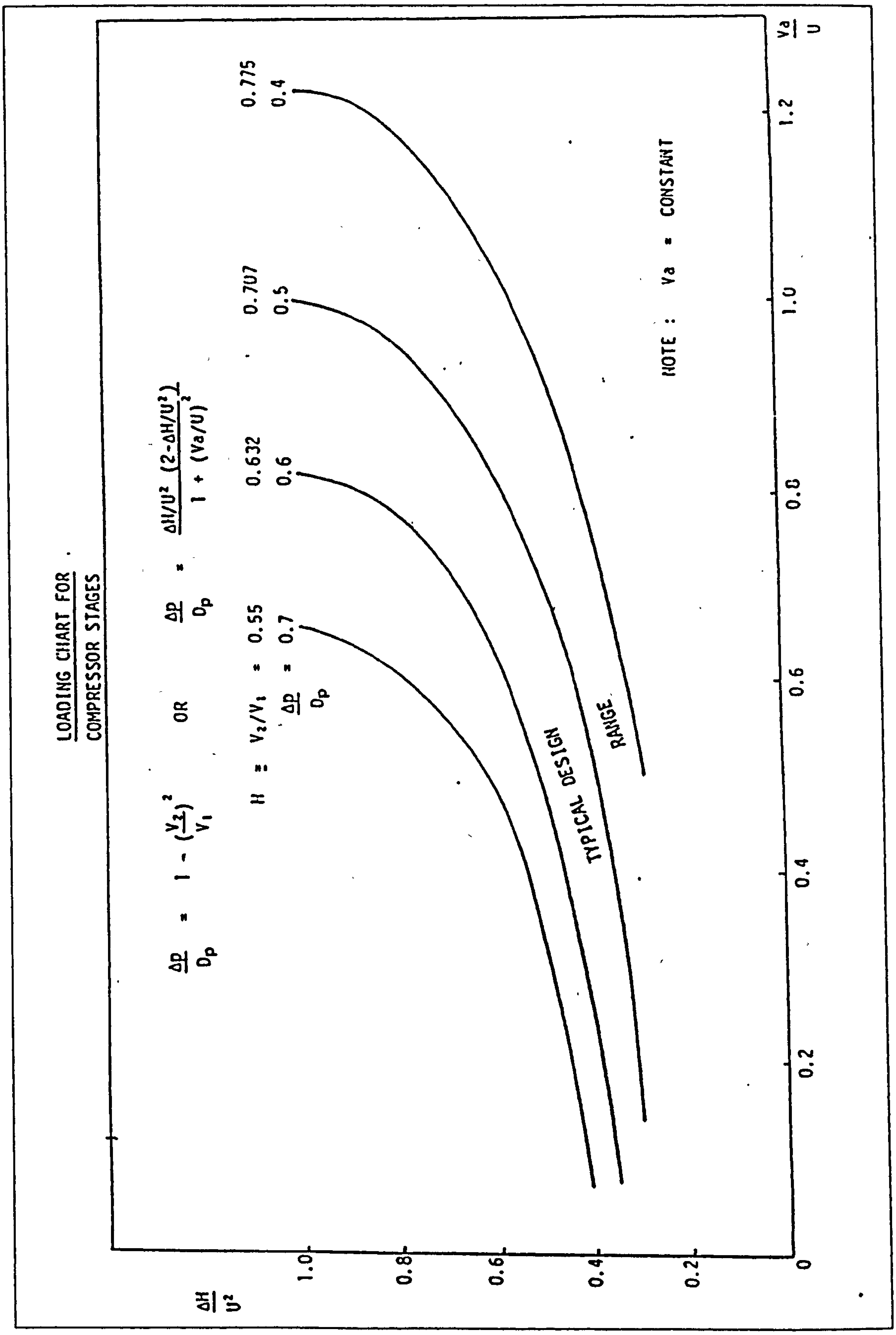


FIGURE 4.2 Efficiency Versus Loading And Flow Coefficients

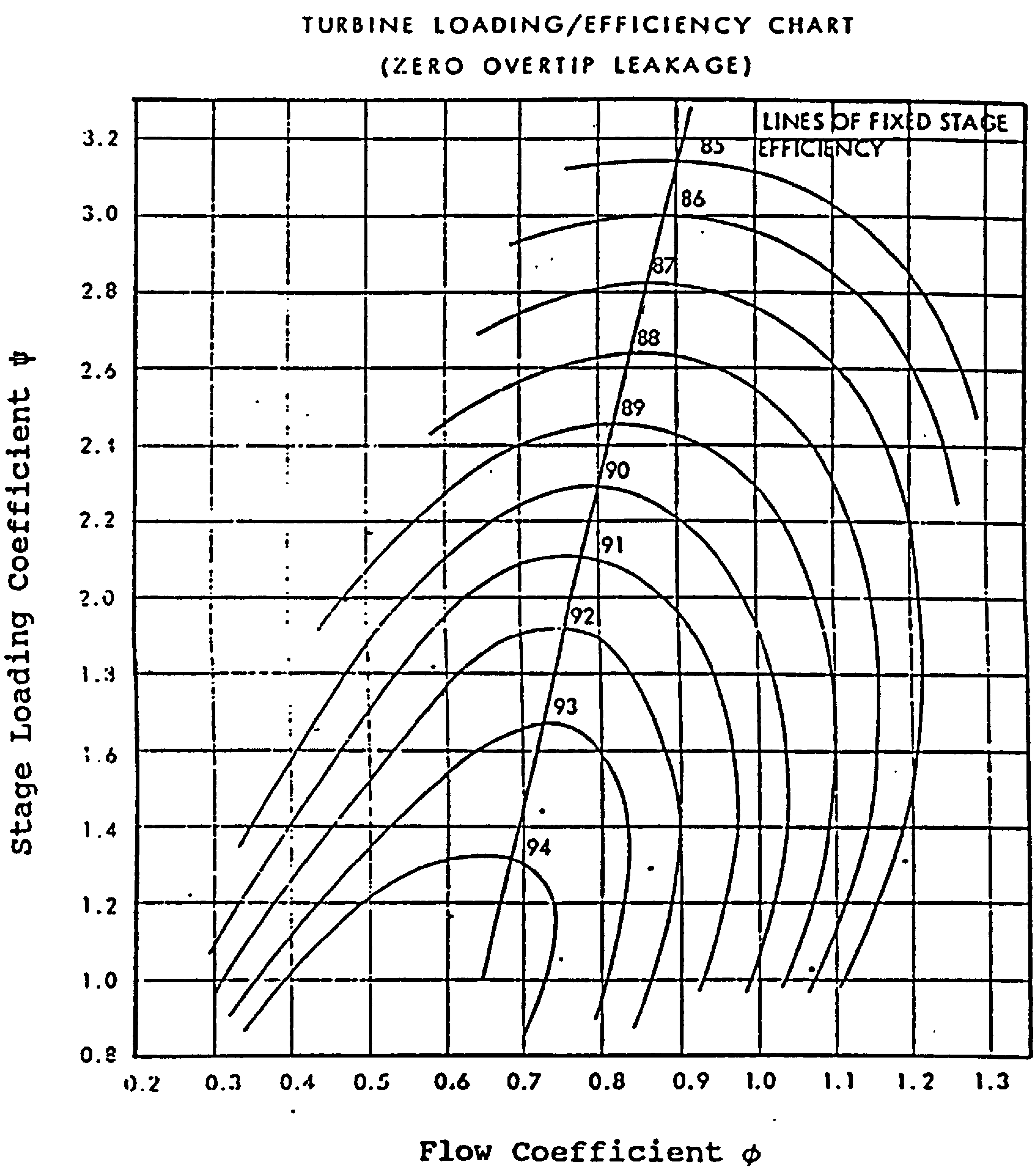
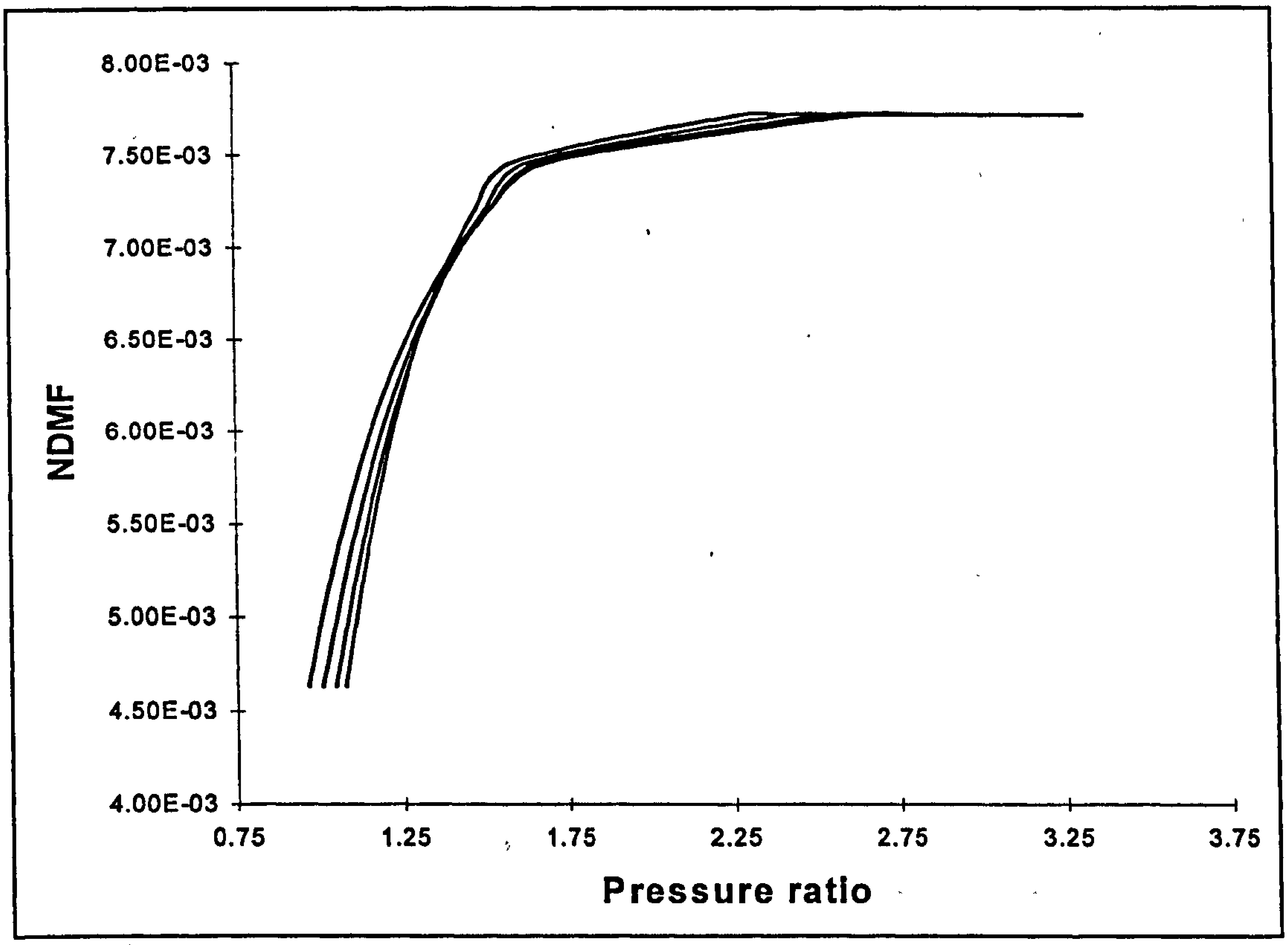


FIGURE 4.3 HP. Turbine Characteristics



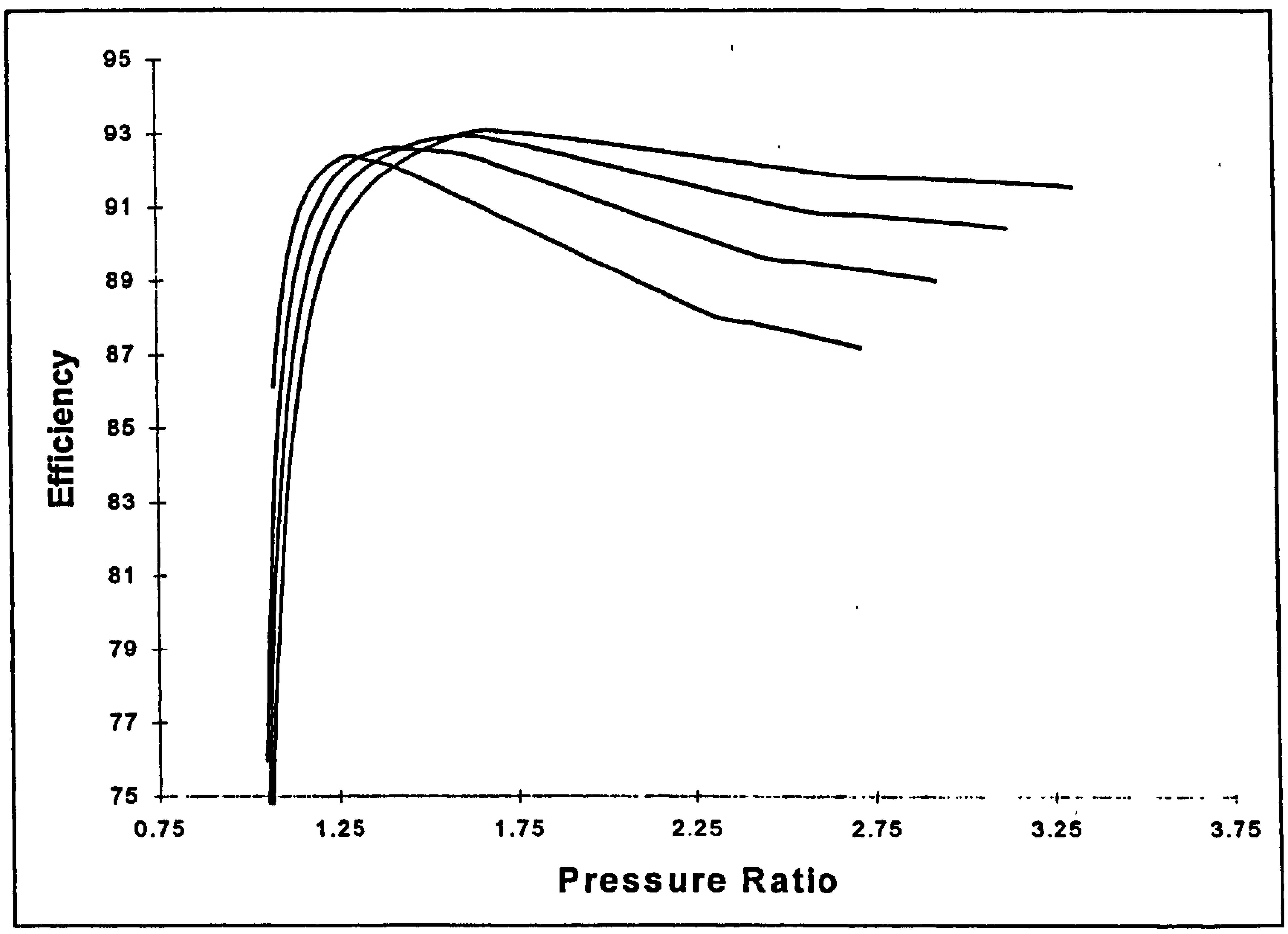
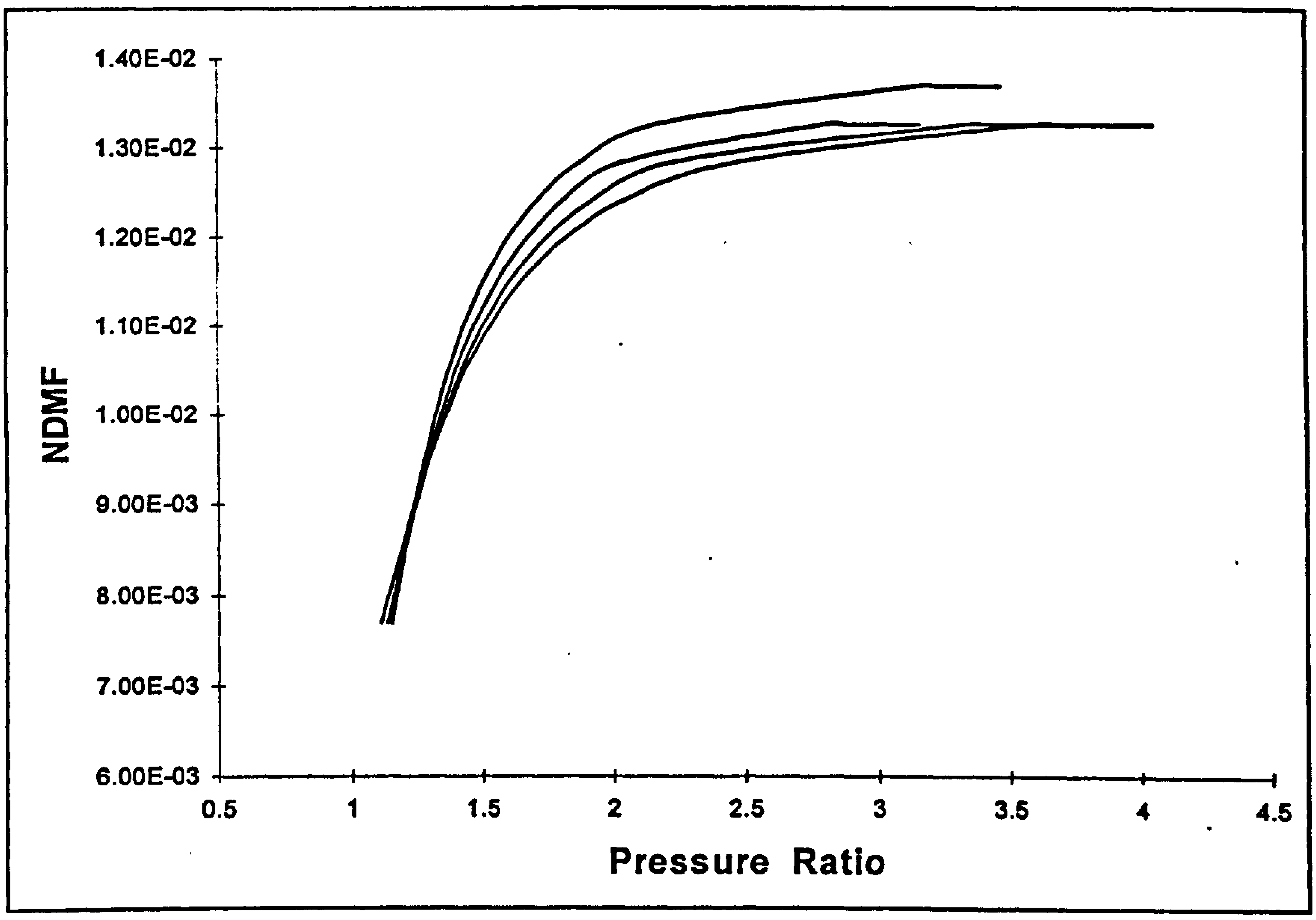


FIGURE 4.4 LP Turbine Characteristics



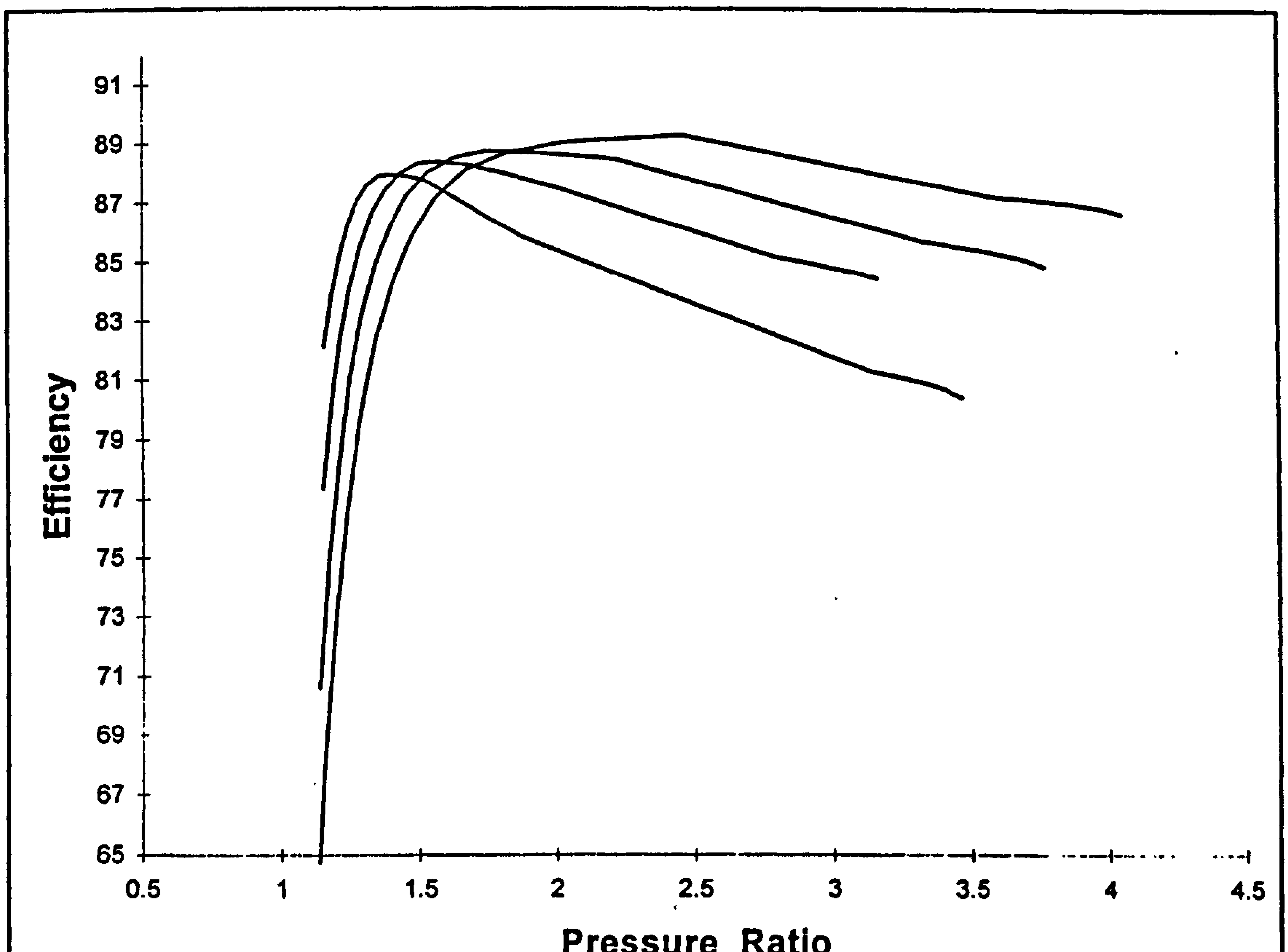


FIGURE 4.5 TFTJ - LPC Running Line

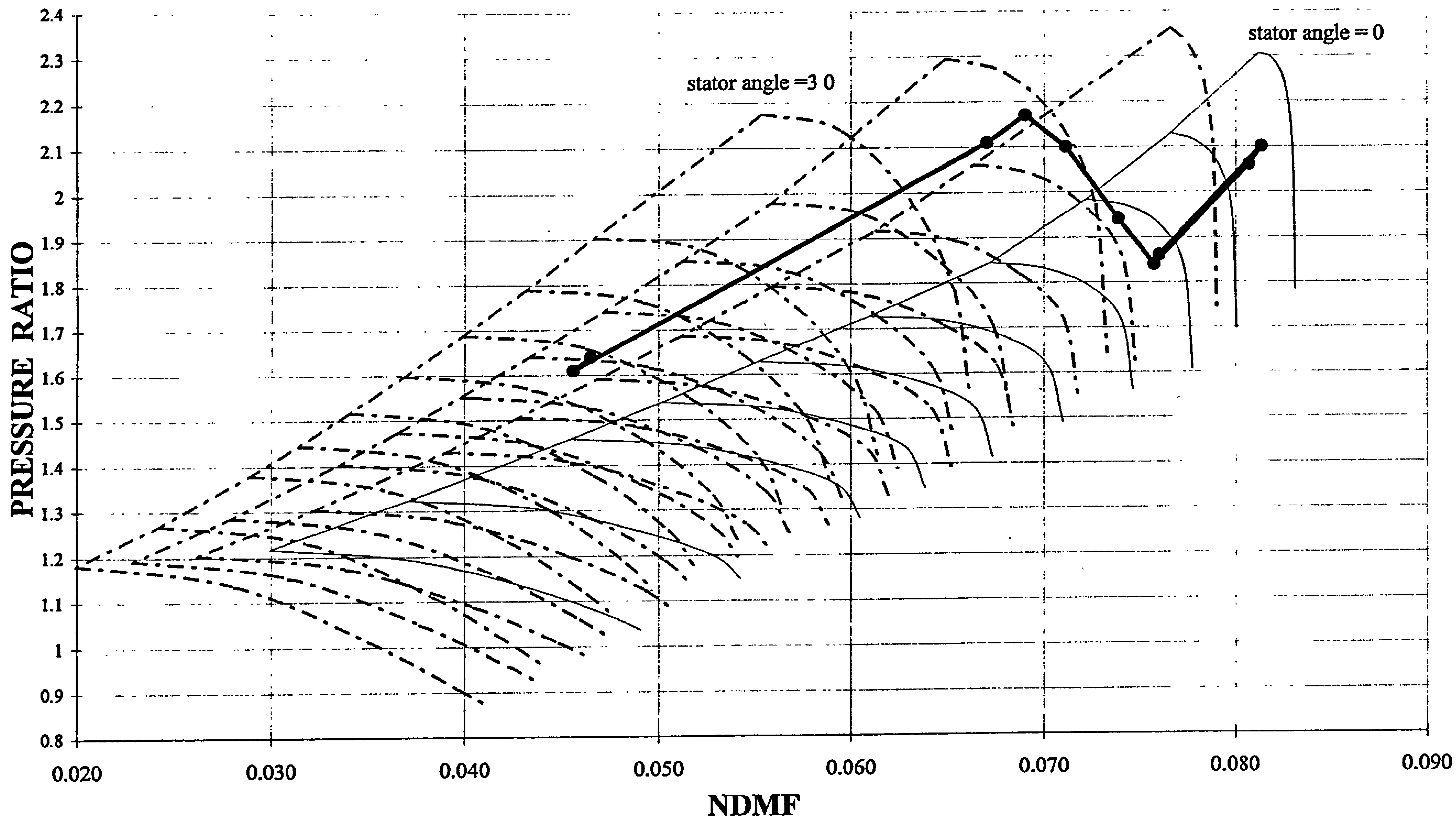


FIGURE 4.6 TFTJ - IPC Running Line

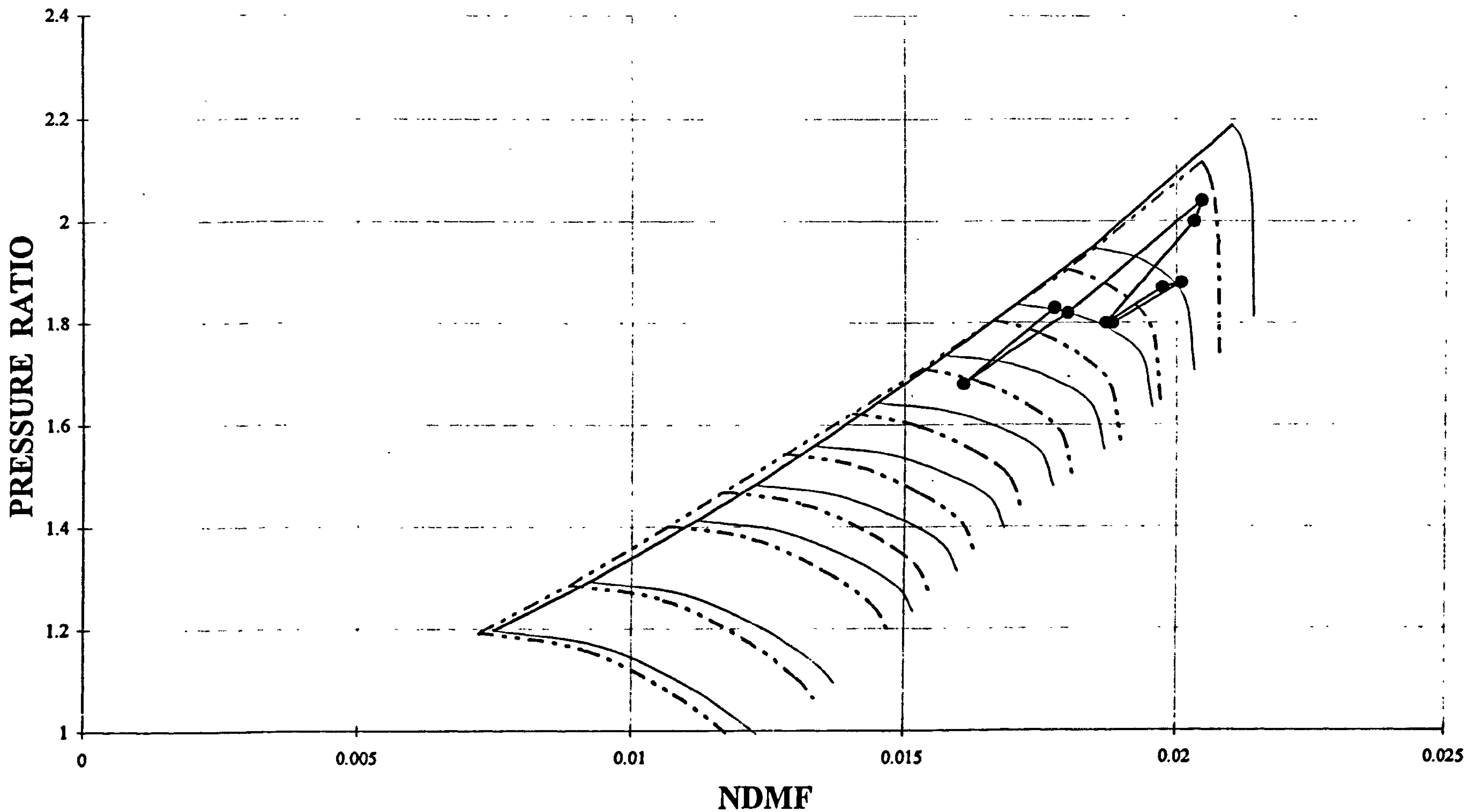


FIGURE 4.7 TFTJ - HPC Running Line

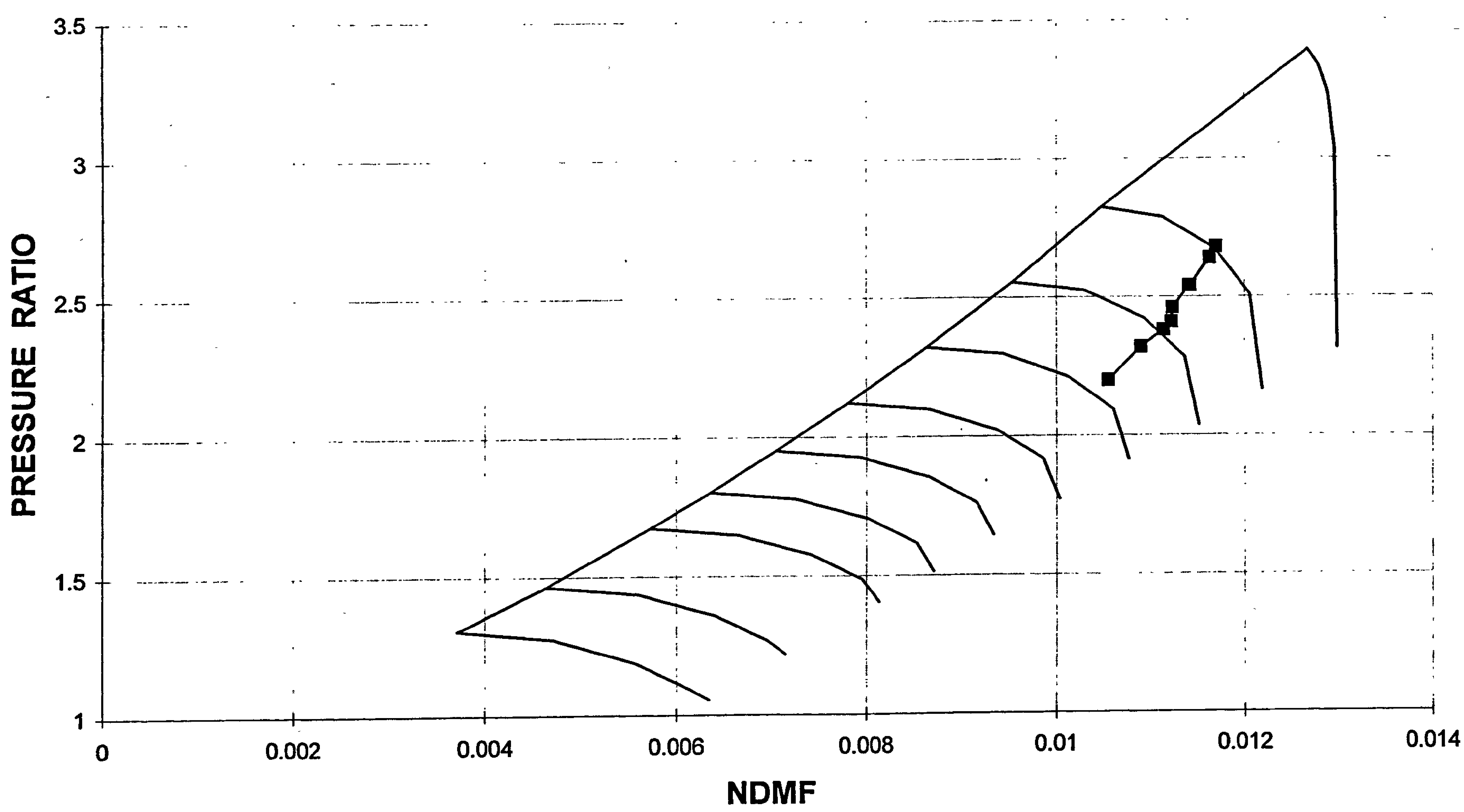


FIGURE 4.8 MTF - LPC Running Line

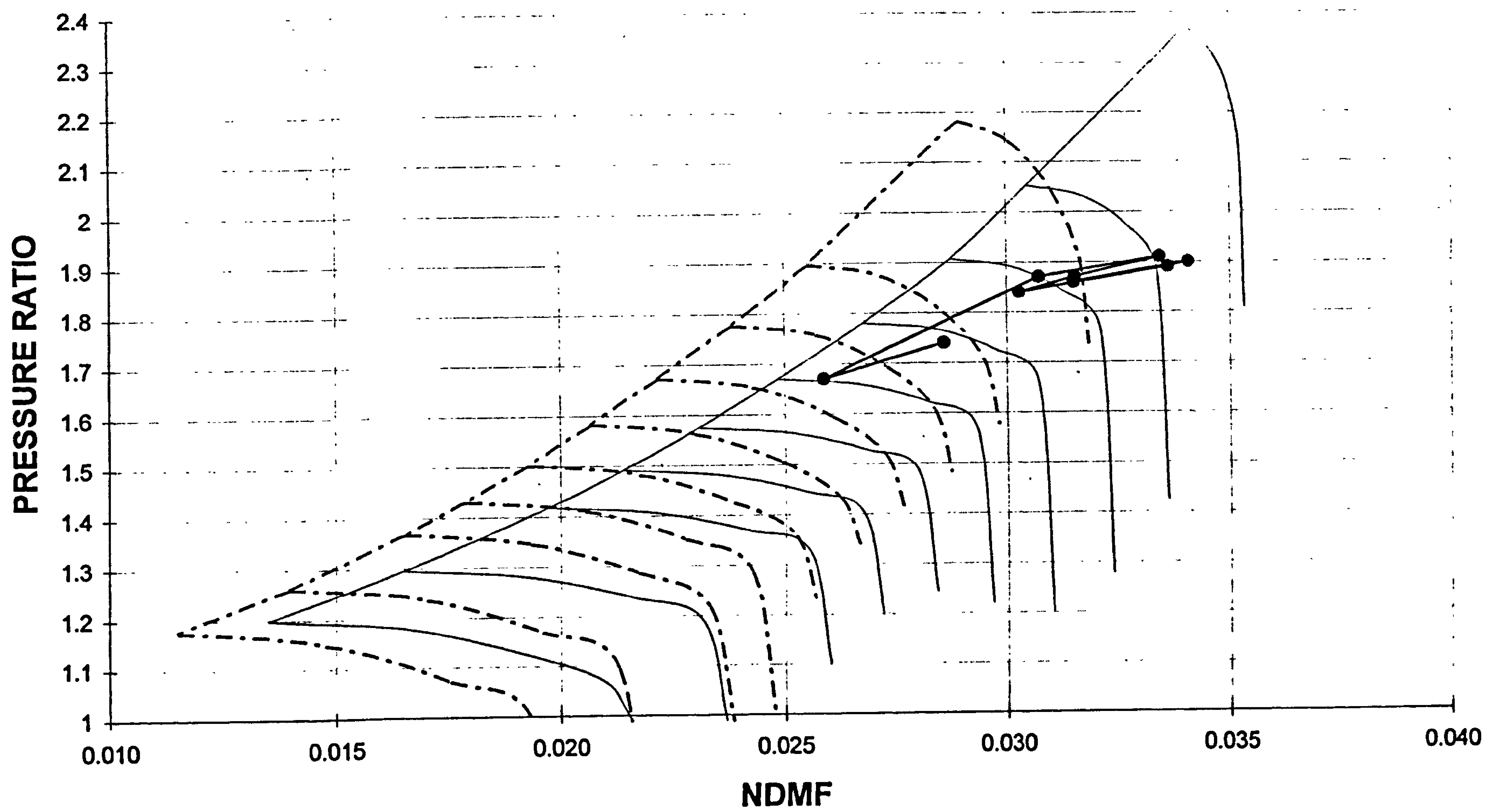


FIGURE 4.9 MTF - FAN Running Line

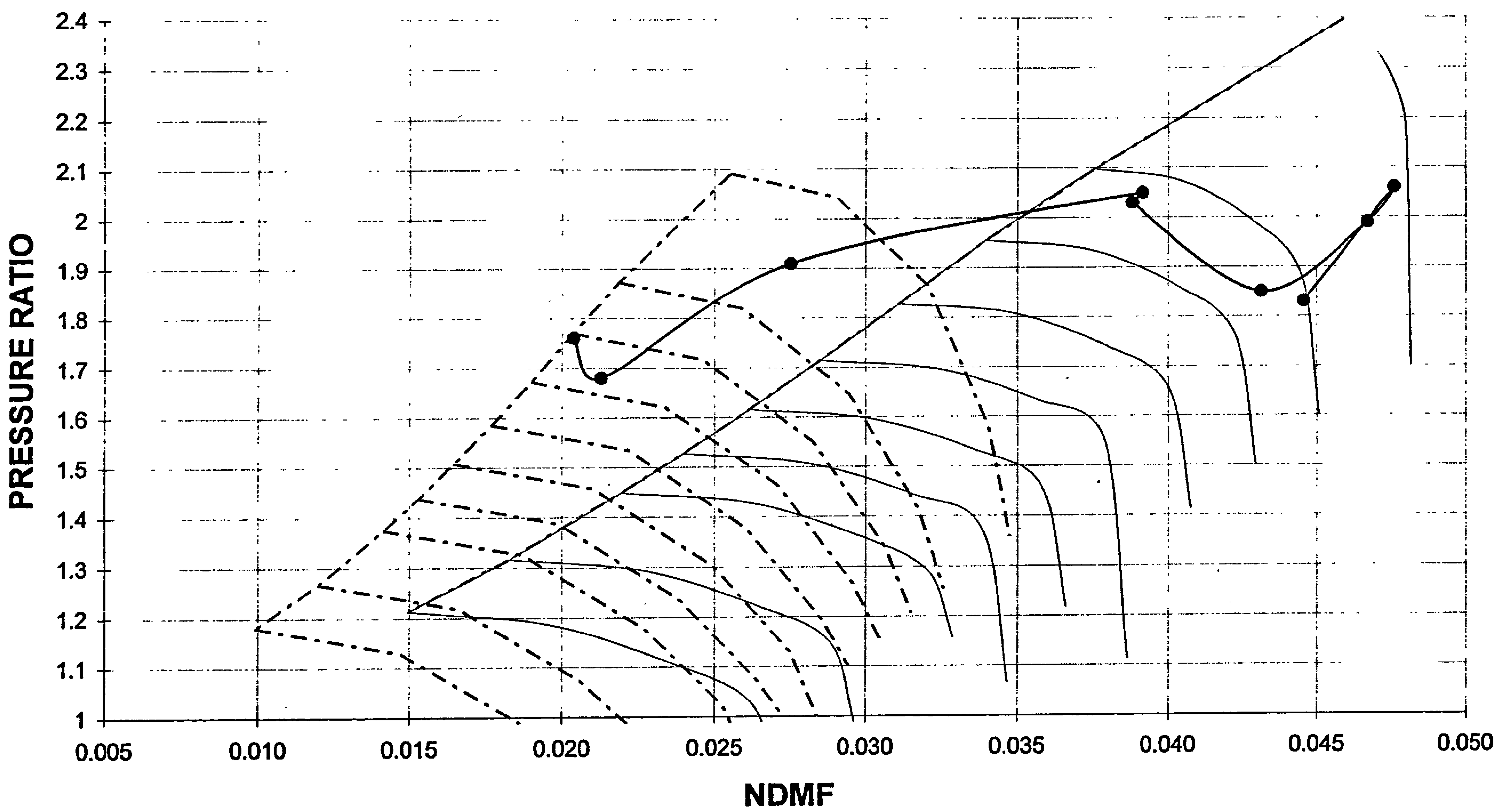


FIGURE 4.10 MTF - IPC Running Line

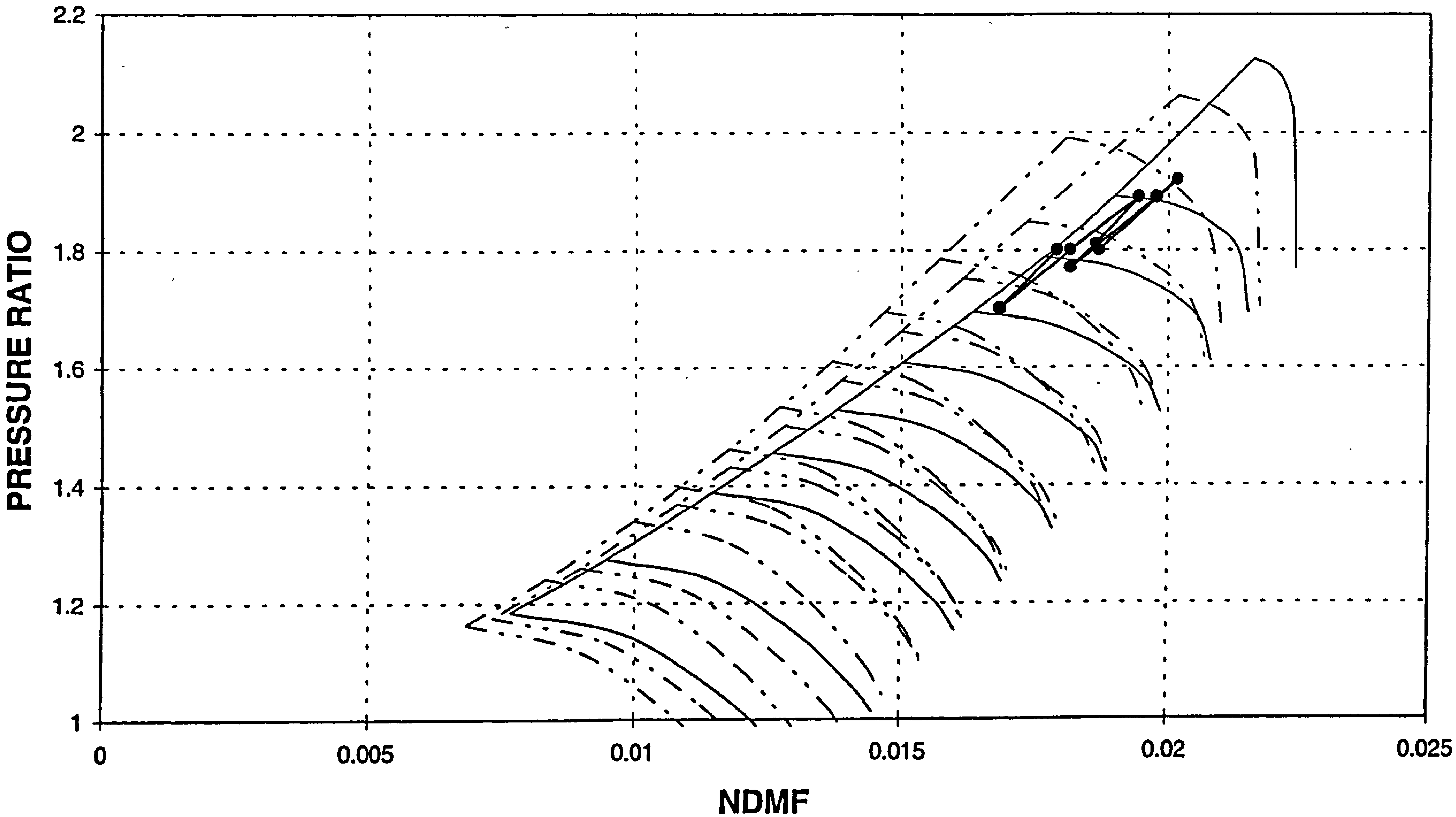


FIGURE 4.11 MTF - HPC Running Line

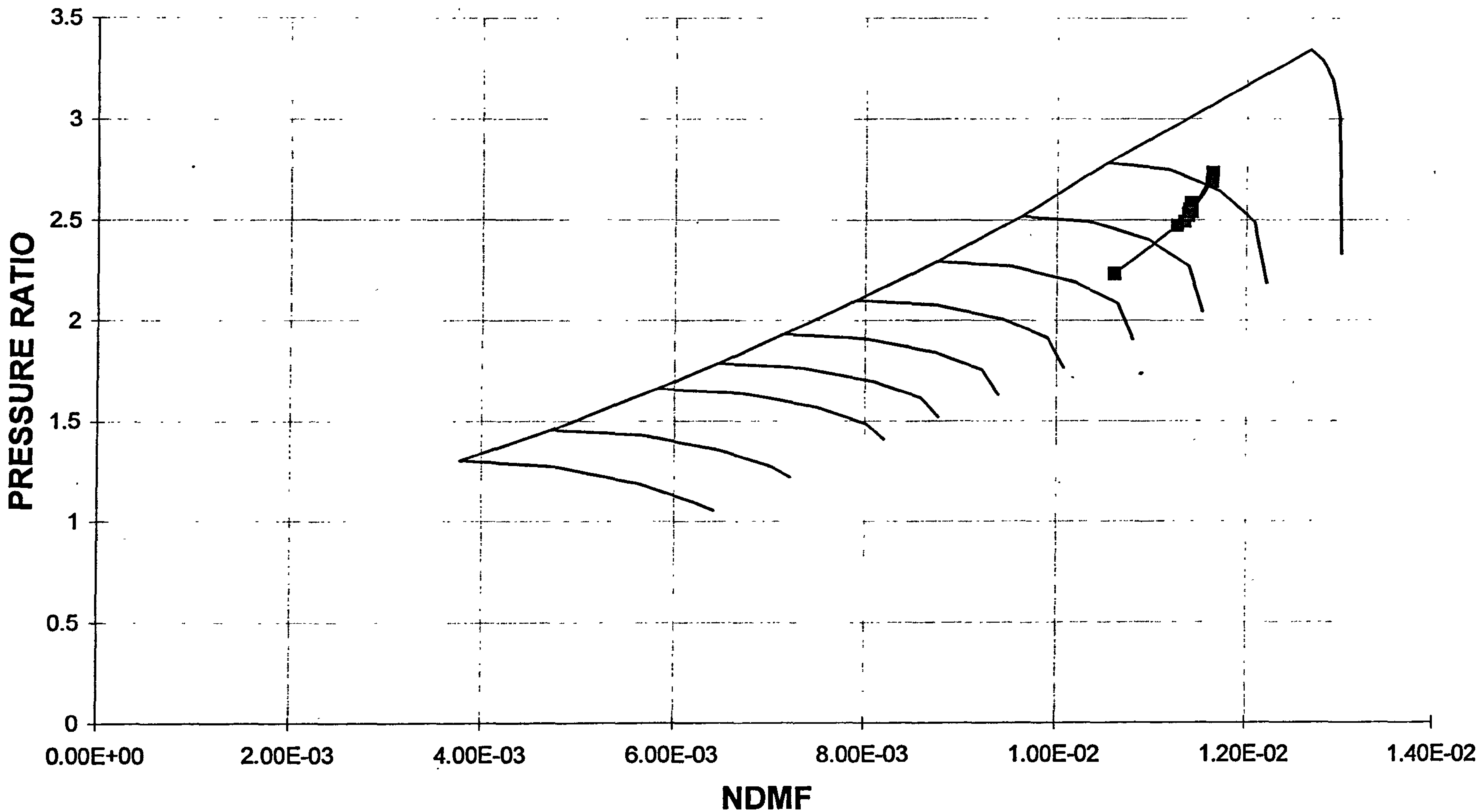
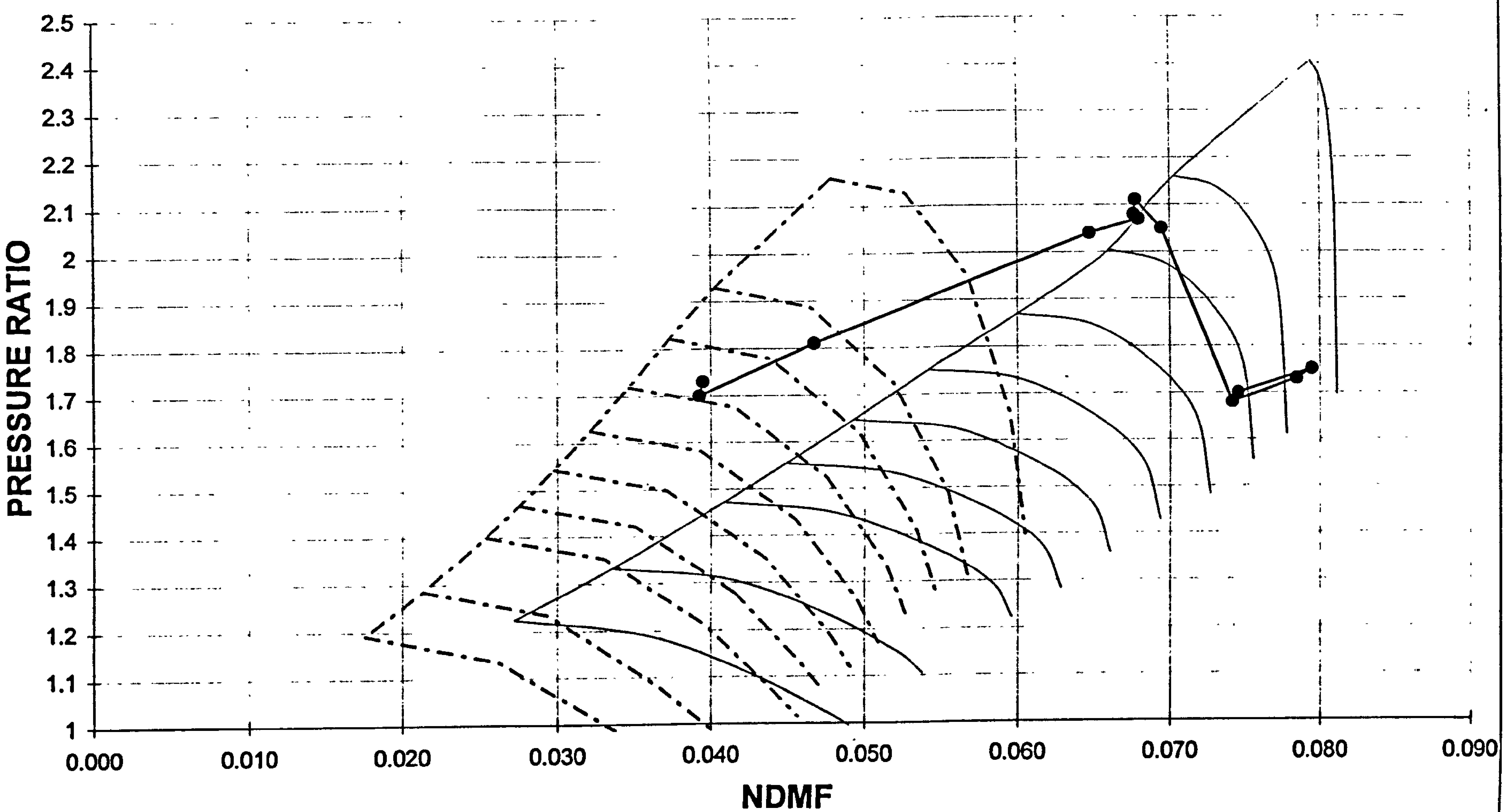


FIGURE 4.12 DBE - LPC Running Line



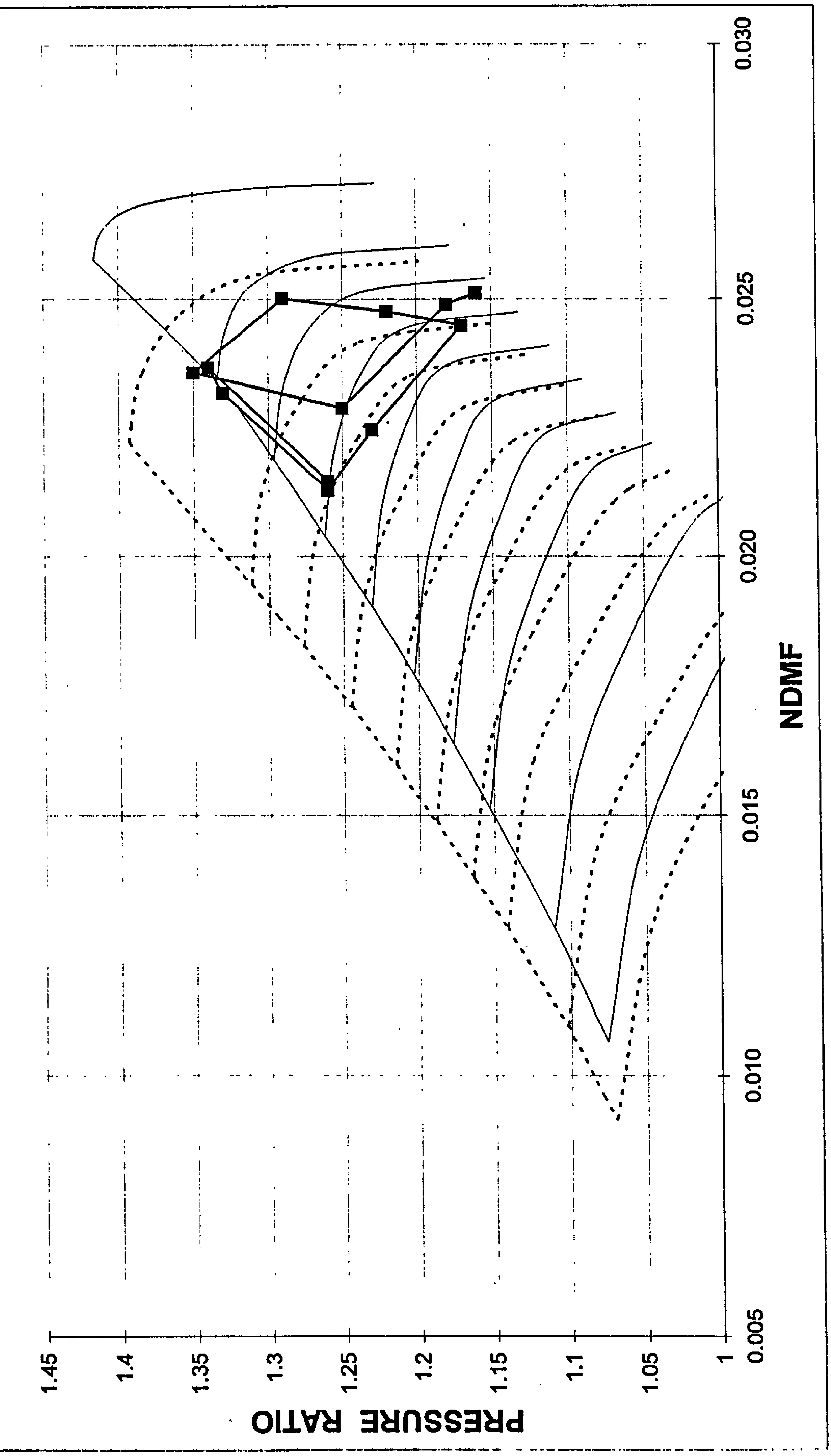


FIGURE 4.13 DBE - IPC Running Line

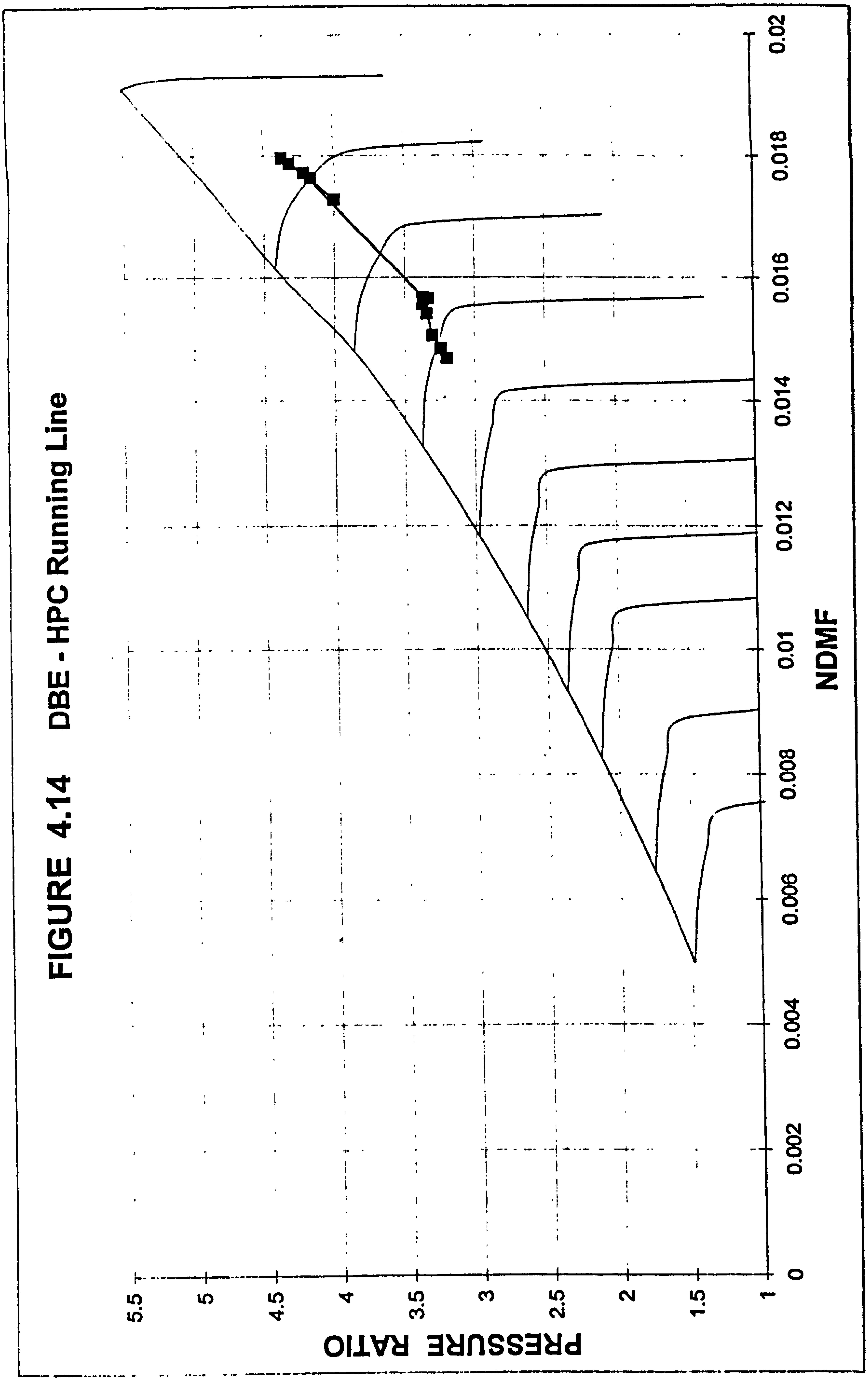
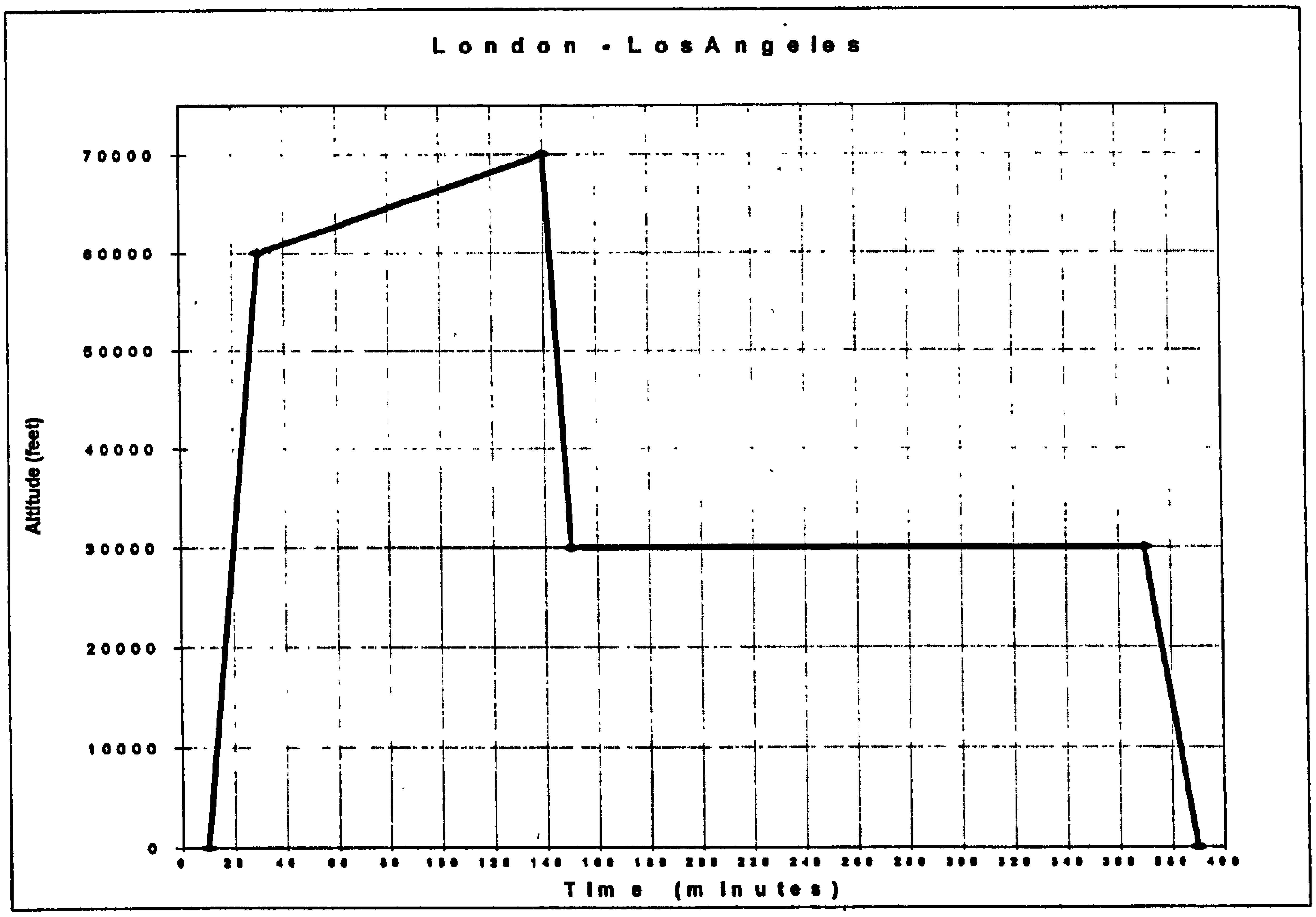
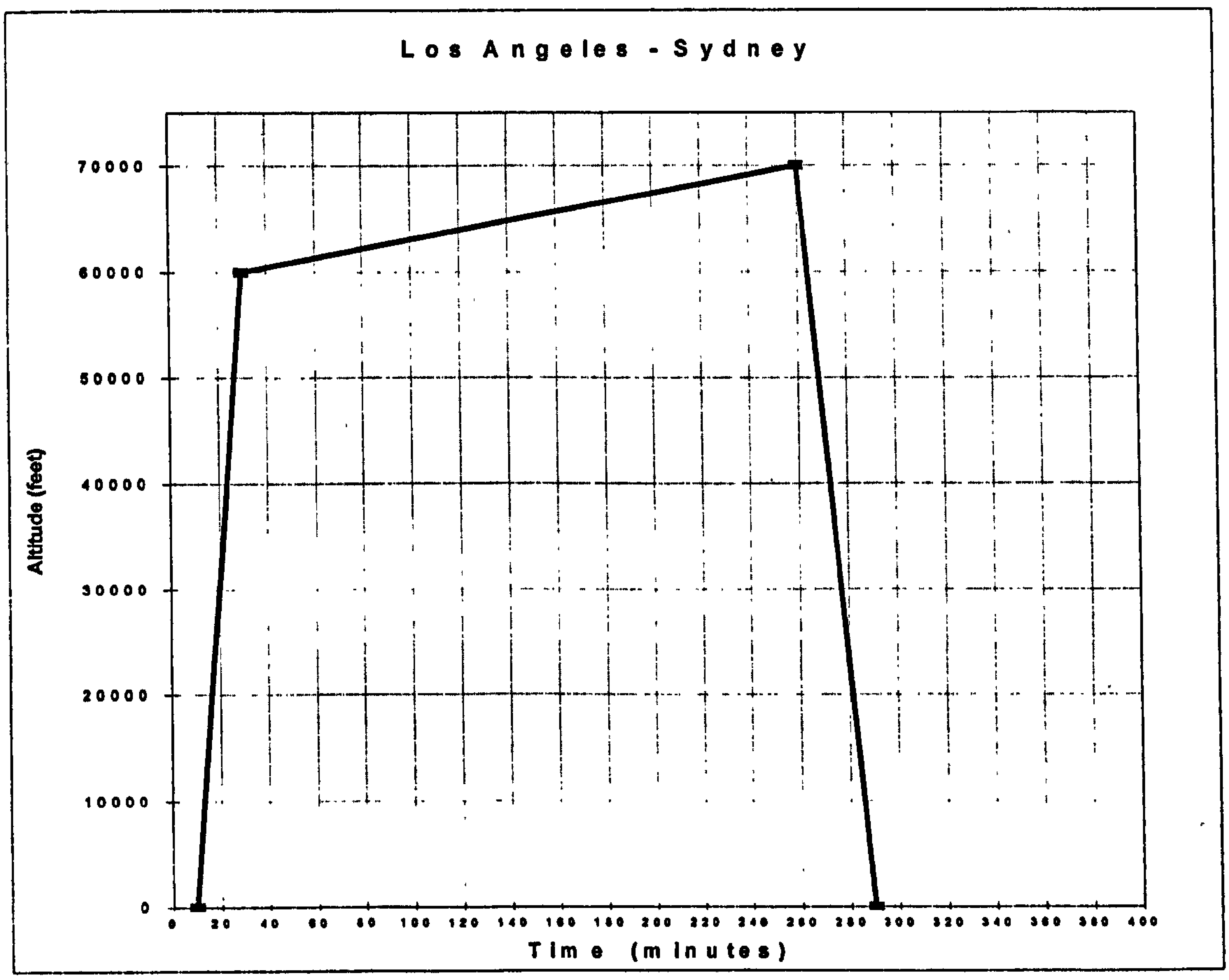


FIGURE 4.14 DBE - HPC Running Line

FIGURE 4.15 Mission Profiles.





CHAPTER 5

Airframe Engine Integration

In this chapter we are concerned with the calculation of the net propulsive force of a simple isolated or off-wing nacelle. That will include, firstly, the calculation of the drag due to air friction on the external surface of each nacelle of the three engines. Secondly, it will also calculate the pre-entry drag due to the intake and its air flow ratio, as well as the afterbody drag due to the nozzle and its performance, together with the wave drag due to the shock waves. A sizing calculation will be carried out for the whole nacelle including the intake and the nozzle.

CHAPTER 5

Airframe Engine Integration

5.1 Introduction

The integration of a turbojet or a turbofan engine into an airframe is a complex engineering issue which is multidisciplinary in nature. It is concerned with the aerothermodynamic, aeromechanical systems and operational aspects of the installation. This comprises the engine, air intake, exhaust system, and the nacelle or housing surrounding the engine.

In this chapter we are concerned with the calculation of the net propulsive force of a simple isolated or off-wing nacelle. That will include the calculation of the drag due to the friction of air on the external surface of each nacelle of the three engines, and the pre-entry drag due to the intake and its air flow ratio, the afterbody drag due to the nozzle and its performance and finally the wave drag due to the shock wave. A sizing calculation will be done for the whole nacelle including the intake and the nozzle.

The total drag of the nacelle at supersonic cruise and subsonic cruise will be obtained and SFC calculated in the previous chapters will be adjusted in order to obtain a better comparison between the three engines. In fact, this will show the effect of geometry on the installed thrust.

A major aim of this chapter is to obtain the thrust corrections due to the forbody and the afterbody drag (intake and nozzle drag), together with the external drag of the nacelle (friction drag). The throttle dependent drag at climb and acceleration and the effect of the engine spillage on the wing performance are not included in this chapter due to the complexity and the lack of information on this subject.

5.2 Thrust And Drag Definition

Figure 5.1 shows an isolated single-stream nacelle with the different forces acting on it. It can be seen by applying Newton's laws between planes located at infinity upstream and downstream of the nacelle in real flow that:

Thrust is the summation of the forces acting on the internal surfaces of the engine nacelle and pre-entry and post-exit streamtubes from minus to plus infinity. The fully-expanded net thrust is given by:

$$F_{N'} = F_{G,00} - F_{G,0}$$

Here, the term, $F_{G,00}$ is the ideal fully-expanded gross thrust of the engine, and $F_{G,0}$ is the intake momentum drag of the engine air. This gives the fundamental definition of the net thrust of the engine. The conventional or standard definition of net thrust, F_N, is given by

$$F_N = F_{G,9} - F_{G,0}$$

Where $F_{G,9}$ is the gross thrust of the engine defined at the geometrical nozzle-exit plane of the engine and $F_{G,0}$ is the intake momentum (ram) drag of the air massflow of the engine $W.V_0$. This definition is the one which is normally quoted by the engine manufacturers. It involves only the engine quantities at a given flight speed V_0 and so can be evaluated independently of the installation of the engine (as $F_{G,1}$, for example, cannot because it depends on the size of the intake of the powerplant). Consequently , F_N can be derived from the measurements of gross thrust and airflow of the uninstalled engine in the sea-level test cell and in the altitude test cell. The gross thrust, $F_{G,9}$ is the gauge stream force of the engine exhaust air at the nozzle-exit plane :

$$F_{G,9} = (W.V)_9 + (P_9 - P_a).A_9$$

The actual value of the term depends on the pressure ratio of the engine, the type of the exhaust nozzle employed in the engine design and the pressure of the air immediately external to the nozzle. The static pressure at the rear of an unducted body can be above, below or equal to the ambient pressure. This is also true for the ducted body or nacelle, but in this case the jet-exhaust air interacts with the surrounding air to modify the local exhaust pressure. If the engine flow at the final nozzle of the engine is sonic or supersonic then the nozzle pressure, P_9 , is independent of the environmental pressure.

A clear distinction must be made between the nozzle environmental pressure or nozzle "base" static pressure, P_b , the nozzle-exhaust static pressure, P_9 , and the free-stream static pressure or ambient pressure, P_a . The values of these pressures dictate the shape of the engine exhaust plume and hence the post-exit stream tube and post-exit force.

Drag is the summation of the forces acting on the external surfaces of the engine nacelle and pre-entry and post-exit streamtubes from plus to minus infinity. The drag ,D , is given by:

$$D = \phi_{pre} + \phi_{nacelle} + \phi_{post}$$

$\phi_{pre} = F_{G,1} - F_{G,0}$
$\phi_{nacelle}$ = Friction force and suction force on the nacelle.
$\phi_{post} = F_{G,00} - F_{G,9}$

The adoption of standard net thrust as the basis of most, if not all, thrust/drag performance accounting systems leads necessarily to the need to account for:

a) The pre-entry drag and the spillage drag of the intake.
b) The post-exit thrust and jet-interference force of the nozzle/afterbody.

The intake-drag terms arise as a consequence of the adoption of the intake momentum "drag" (ram-drag) as the stream force ahead of the propulsion system. The post-exit thrust and force terms arises as a consequence of the adoption of the gross thrust term, $F_{G,9}$, rather than the fully-expanded thrust as the stream force at infinity downstream of the propulsion system. The intake spillage is a result of the failure of the intake cowl to fully balance or recover the pre-entry force. The jet-interference force of the nacelle is a result of the partial recovery of the post-exit thrust of the engine exhaust flow on the external surface of the afterbody of the propulsion system. The intake drag and afterbody/nozzle force terms introduced above are called throttle dependent drags and forces because their magnitudes depend on the operating flow condition of the engine.

5.3 Nacelle Sizing

In order to calculate the powerplant drag we need to define the geometry of the whole nacelle (intake, engine and nozzle). The engine dimensions have been calculated in the previous chapter. In this chapter the intake and nozzle dimensions will be defined. The final aim is to design a standard intake and nozzle which will allow us to compare between the three engines. These standard models will be very close to the real intake and nozzle and this approximation will be enough for the scope of this thesis.

5.3.1 Intake Design And Sizing

The intake is a critical component of the propulsion system for the high-speed aircraft. At cruising speed the inlet must decelerate the air from the airspeed of the plane, which is 2.7 in our case, to about Mach 0.6 relative to the compressor face. In order to be effective this deceleration must take place without generating large transverse velocity components, flow distortions, and total pressure losses. The inlet can be thought of as two separate flow components in series, the supersonic entrance and the subsonic diffuser.

There are two general types of entrance geometries: axisymmetric and rectangular. Both types are candidates for future SST. An axisymmetric intake is, in general, shorter and its pressure recovery is higher, but its installation drag is difficult to predict. Reference 40 gives an analytic and experimental study of a mixed compression axisymmetric inlet at Mach 2.65.

In this work, a rectangular entrance is being studied for use in the high speed civil transport because it is generally easier to control than an axisymmetric entrance (Ref. 37). and its installation drag is easier to predict. Furthermore, the two supersonic civil aircrafts Concorde and TU 144 use rectangular intakes.

There are three sorts of rectangular entrance: external, internal and mixed compression inlets. At flight Mach numbers above 2.5, the mixed compression inlet is used to obtain an acceptable total pressure ratio while obtaining acceptable cowl drag (Ref. 37). This mixed compression inlet will be used for the present design. The typical mixed compression inlet achieves compression through the external oblique shocks, the internal reflected oblique shocks, and the terminal normal shock. The ideal location of the normal shock is just downstream of the inlet throat.

Because of the high cruising speed, each engine inlet must be fairly complex. The inlet geometry must be variable in order to pass the proper amount of air into the engine at conditions ranging from standing still on the runway to supersonic cruise. A much larger inlet area is required to allow sufficient airflow to the engine during supersonic flight than during subsonic flight. Another reason why the inlet must have variable geometry is to avoid unstart. Unstart occurs when, during supersonic flight, the normal shock moves upstream of the geometric throat of the inlet and out the front of the inlet. This can cause compressor stall with consequent loss of thrust. The airspeed of the flow entering the normal shock is about 1.3 This will give a good trade-off between pressure recovery and shock stability. A higher Mach number would reduce total pressure recovery. A lower Mach number would make maintaining the position of the shock in the throat difficult (Ref. 39).

Reference 41 gives the characteristic design study of a mixed compression two-dimensional inlets. Figure 5.2 presents the characteristic solution for the basic inlet at the design Mach number of 2.7 . The inlet wedge was comprised of two ramps with initial compression angles of 5 °each. The second oblique shock, originating at the junction of the first and second ramp surfaces (x = 0.67), was located such that it intersected the cowl lip, while the initial ramp shock was located forward of this point. This provided supersonic spillage of 0.5 % of the maximum capture mass flow. The inlet has a 0° internal cowl lip angle. The resulting cowl lip shock was subsequently cancelled at the ramp shoulder point, x = 3.09. Downstream of the cowl oblique shock, the flow was compressed isentropically to a throat Mach number of 1.3. The flow angle at the throat had a nominal value of -4°. Theoretical total pressure recovery behind the terminal shock in the throat (x = 3.75) was 0.953. Table 5.1 shows the intake oblique shocks parameters.

θ	β	Mach Number	P	η_t
5	25.51	2.7 / 2.472	1.5313	0.9959
5	27.86	2.472 / 2.243	1.525	0.9964
10	35.18	2.243 / 1.857	1.519	0.9807
isentropic compression		1.857 / 1.3	1.519	1
Normal shock		1.3 / 0.786	1.4901	0.9794
		0.786	1.459	0.953

Table 5.1 Oblique Shocks Parameters (Ref. 41)

5.3.1.1 Intake Size

The aim of this paragraph is to calculate the inlet geometry at the supersonic design point. That will include the entrance area $A_1 = h_1 * w_1$; the throat area $A_t = h_t * w_t$ and the total length of intake ($L_{total} = L_{sup} + L_{sub}$) including the supersonic entrance and the subsonic diffuser. Inlet design and sizing begin with an analysis of engine airflow requirements. Figure 5.3 shows the ratio of free stream captured area required by the engine airflow to the design area in function of flight Mach number M_0 (obtained from Turbomatch simulations).

5.3.1.2 Entrance Area A_1

The future supersonic intake, which is intended for efficient long-rang flight, includes the provision of an internal boundary layer bleed. The bleed air is discharged through nozzles located on the surface of the nacelle. In this case the total intake mass flow is shared between engine flow and bleed flow, say:

$$m_{intake} = m_{engine} + m_{bleed}$$

The ratio of bleed flow to engine flow is a function of many parameters and the determination of its value is a very complex task, which is outside the scope of this project. However this value can be limited between 4% to 12% for future supersonic civil intake (Ref. 37). A value of 10 % will be used for the present design.

The design point (M_0=2.7) free stream area required to pass the above intake mass flow is:

$$A_0 = (m_{intake} * \sqrt{T_0}) / (Q * P_0)$$

Where Q is the mass flow parameter. The nacelle maximum external diameter D_{max} is equal to the LPC diameter plus 0.15 m for the nacelle structure(Ref 5). The LPC diameter is given in the previous chapter for the three engines.

At supersonic design point A_0 is equal to the inlet area A_1. There is no spillage at this point and the spillage drag is kept to a minimum value. The geometry of the inlet is now defined, the height is $h_1 = D_{max} - 0.075$ m and the width is $w_1 = A_1 / h_1$.

5.3.1.3 Throat Area A_t

The calculation of the throat area requires knowledge of the shock pressure recovery η_s. Turbomatch program uses a total pressure recovery given by military specification MIL-E-5008B (Ref. 38), where:

$$\eta_t = 1 - 0.075*(M_0 - 1)^{1.35}$$

The total pressure recovery η_t includes the shock pressure recovery η_s plus the subsonic duct pressure recovery η_D : ($\eta_t = \eta_s * \eta_D$). η_s reflects the pressure loss due to the shock waves and η_D reflects the pressure loss due to the subsonic diffusion.

From Table 1 η_s can be calculated, $\eta_s = 0.953$, and the subsonic duct pressure recovery η_D is estimated to be around 0.98. That would give a total pressure recovery of 0.934 verses a value of 0.8465 calculated from the above formula.

The conservation of mass flow between the inlet entrance A_1 and the throat area A_t (we assume that bleed takes place after the throat) gives:

$$m_o = m_1 = m_t \quad \text{or} \quad \rho_o V_o A_o = \rho_1 V_1 A_1 = \rho_t V_t A_t$$
$$A_1/A_t = (\rho_t V_t)/(\rho_1 V_1)$$

And for adiabatic flow in the inlet ($T_1 = T_o = T_t$), the above area ratio can be written in terms of total pressure and mass flow parameter as:

$$A_1/A_t = (P_t / P_o) * (Q_t / Q_o)$$

P_o, Q_o can be found for Mach 2.7 and an altitude of 18,750 m. $P_t = \eta_s * P_o$ and Q_t can be found for the subsonic throat Mach number. So A_t can be calculated from the above formula. The width of the throat equal to inlet width , ($W_t = W_1$) and the throat height is $h_t = A_1 / W_1$. Figure 5.4 shows the layout of the different intake inlet areas for the three engines.

5.3.1.4 Inlet Length

The calculation of the total length is divided into two parts, the supersonic entrance length and the subsonic diffuser length. A coordinate system (X,Y) is defined in Figure 5.2.

A- Supersonic Entrance Length L_{sup}

The length of the supersonic entrance is a function of the shock angles. Decreasing the shock angles would improve the pressure recovery, but would also increase the intake length. The intake supersonic entrance length extends from the upper lip of the cowl to the throat, in our case the throat is where the normal shock takes place (X = 3.75). In the real intake the normal shock is just after the geometric throat, this approximation is enough for the purpose of this project.

B- Subsonic Diffuser Length L_{sub}

The subsonic diffuser must connect a rectangular throat to an annular compressor inlet. At the same time the diffuser must decelerate the air from about Mach number of 0.8 to about Mach number of 0.5 This reduction must take place with minimal loss in total pressure, minimal total pressure distortion and without producing large transverse velocity components at the face of the compressor.

The determination of the subsonic diffuser length is very complex because of its shape and the lack of detailed information in the literature. Reference 39 suggests a subsonic diffuser for future SST, where the diffuser length will be approximately twice the compressor diameter and this approximation will be used for this project.

$$L_{sub} = 2 * D_{max}$$

Table 2 shows the different lengths for the three engines, and Figure 5.5 shows a general view of the subsonic diffuser.

5.3.2 Nozzle Size

A variable geometry convergent-divergent nozzle will be used for this project. The exit and the throat areas are known at the supersonic design point from Turbomatch simulation (see Chapter 3). The secondary nozzle half angle is estimated to be about 7° and the divergent length can be found (Ref. 38).

An afterburner is used to increase the engine thrust during climb and acceleration. The length needed to install the afterburner is estimated to be about 1.5 m (Ref. 38). Table 5.2 shows the length of the different nozzles and afterburners for the three engines. Figure 5.7 shows the general layout of the nacelle for the three engines.

Intake	TFTJ	MTF	DBE
Lip	4.1392	3.64	4.0976
Shoulder	6.1491	5.4075	6.0873
Throat	7.4625	6.5625	7.3875
L_{sup}	7.4625	6.5625	7.3875
L_{sub}	3.98	3.5	3.94
L_{total} (m)	11.443	10.063	11.328
Engine Length (m)	2.55	2.886	2.904
Nozzle			
Throat area	1.07	1.12	0.73
Exit area	2.7	2.4	2.9
Length (throat to exit)	2.798	2.256	3.208
After Burner Length	1.5	1.5	1.5
Nozzle Total Length	4.298	3.756	4.708
Total Length (m)	18.291	16.705	18.94

Table 5.2 Intakes, Engines And Nozzles Length

The total nacelle length is 18.3 m, 16.7 m, and 18.9 m for the TFTJ, MTF and the DBE respectively. Table 5.3 compares the above results with the two existing supersonic transport aircraft Concorde and TU 144.

	N. Of Passengers	Cruise Mach n.	Engine Diameter	Nacelle Total length	Max. Thrust (kN)
SST	300	2.7	1.6~1.9 m	17~19 m	250 at least
Concorde	100	2.04	1.2 m	13 m	177
TU - 144	140	2.34	1.35 m	22 m	215

Table 5.3 Nacelle Supersonic Aircraft Comparison

5.4 Drag Calculation

After sizing the engine, the main drag components will be calculated for the three engines at supersonic and subsonic points. The nacelle total drag will include the skin friction drag, pre-entry drag, form drag or pressure drag for intake and nozzle, and the wave drag for supersonic speed.

5.4.1 Friction Drag

The supersonic drag calculation is based on theoretical and numerical methods. The skin friction drag components are similar in both supersonic and subsonic flow. The effects of compressibility and heat transfer must be accounted for at supersonic speeds. The drag of the nacelle due to friction and form is computed as:

$$D = C_f * FF * IF * q * S$$

Where C_f is the compressible flat-plate skin friction coefficient, FF is the nacelle form factor, q is the dynamic head pressure ($q = 1/2 * \rho_o * V_o^2$), and S is the nacelle wetted area. The flat-plate, compressible, turbulent, skin friction coefficient is determined from the equation given in Reference 36:

$$C_f = t * f^2 * 0.43 / (\log_{10}(R_{NL} * t^{1.67} . f)^{2.56})$$

Where:

$$t = [1 + 0.178 * M_o^2]^{-1}$$
$$f = 1 + 0.03916 * M_o^2 * t$$

The Reynolds number, R_{NL}, is based on the component length and is given by (Ref 2):

$$R_{NL} = (7.086 * M_o * [1 - (1.98 * H/288)]^{4.016} * 10^6) * L \qquad \text{If } H < 36.089$$
$$R_{NL} = (12.77 * M_o * EXP(-H/20.81) * 10^6) * L \qquad \text{If } H > 36.089$$

Where H is the altitude in thousands of feet.

The form factor FF for the nacelle is give by (Ref. 36):

$$FF = 1 + 0.35/FR$$
$$FR = \text{component length} / (\text{Width} * \text{Height})^{1/2}$$
IF = interference factor = 1.5 for nacelle mounted flush to the wing

5.4.2 Pre-Entry Drag

The pre-entry drag is due to the fact that the free stream capture area is less than the intake geometric area i.e. when the area ratio A_o / A_e is less than unity. The static pressure is decreased at the intake entrance due to the expansion of the free stream tube.

This would create a positive force in the drag sense so long as the flow ratio is less than 1. This force is given by:

$$D_{pre} = (P_e - P_o) * (A_e - A_o)$$

The pre-entry drag is a function of the free stream mach number, mass flow and the intake geometrical area. It is zero when the flow ratio is 1, and increases when the flow ratio decreases. Figure 5.2 shows the variation of the free stream captured area relative to the supersonic design area.

5.4.3 Wave Drag

The wave drag is introduced only at supersonic speed. It is due to the increase in the static pressure through the shock wave and it is given by:

$$D_{wave} = (P_w - P_0) * [(A_e - A_o) + (A_{max} - A_e)]$$

The wave drag is a function of the free stream mach number, wave shock parameters (shock angle, M_2 ...) and the inlet intake geometrical area and the maximum nacelle area.

5.4.4 Calculations And Results

The nacelle is divided into three parts, intake, engine and the nozzle. The skin friction coefficient C_f is the same for the three parts, but the form coefficient is calculated for each of them. The intake and the nozzle were designed primarily for the supersonic design point so that the pre-entry drag is minimal. For the subsonic design point the variable inlet intake area is reduced in order to decrease the pre-entry drag at this point. If the intake geometrical area is much higher than the free stream capture area then the pre-entry drag would be high and unacceptable. A value of 90 % of the flow ratio at this point would give an acceptable pre-entry drag. The reduction of the intake supersonic design area would be around 40%. Figure 5.6 shows a schematic variable intake inlet area. Table 5.4 shows the different components drag at supersonic and subsonic points for the three engines.

SUPERSONIC	TFTJ	MTF	DBE
C_f	0.00145	0.00145	0.00145
FF	1.057/ 1/ 1.151	1.066/ 1/ 1.163	1.054/ 1/ 1.125
Nacelle Drag(N)	FRICTION &	FORM &	INTERFERENCE
INTAKE	4767	3937	4574
ENGINE	1710	1641	1865
NOZZLE	2636	2144	2889
Total Nacelle(N)	5831	4957	6197
Pre-entry drag	48	50	42
WAVE DRAG	610	0	1590
TOTAL (N)	9771	7772	10960

SUBSONIC	TFTJ	MTF	DBE
C_f	0.00183	0.00182	0.00182
FF	1.044/ 1/ 1.19	1.049/ 1/ 1.133	1.044/ 1/ 1.067
Nacelle Drag (N)	FRICTION &	FORM &	INTERFERENCE
INTAKE	2787	2255	2682
ENGINE	1012	955	1104
NOZZLE	1516	1216	1611
TOTAL (N)	5315	4426	5398
Pre-entry drag	1692	1644	1658
TOTAL (N)	7007	6070	7056

Table 5.4 Different Drag Components

5.5 Installed Thrust

The thrust given in the previous chapter was the uninstalled thrust. The aim of this chapter is to estimate the installed thrust for each engine at supersonic and subsonic design points. Once the nacelle drag is estimated, the installed thrust can be calculated, but it will be lower than the net thrust required to maintain the design Mach numbers. In a real engine design, the starting design uninstalled thrust should be around 10% higher than the required thrust, then the nacelle sizes and drag estimation will determine the installed thrust. Many iterations are needed in order to evaluate the final size and nacelle drag in order to obtain the required installed net thrust.

The installed thrust is the engine thrust, given by Turbomatch program, minus the nacelle total drag calculated above. When Turbomatch program calculated the uninstalled engine thrust the intake pressure recovery was estimated by military specification MIL-E-5008B (Ref. 41), where its values was 0.846 at supersonic design point. However, we found that the design theoretical intake pressure recovery is 0.934 at this point which is 10.4 % higher than Turbomatch estimated. This would improve the net thrust approximately by the same amount for the same engine size, and the SFC will improve by around 2% at the supersonic point. Table 5.5 shows the SFCs and fuel flows for both uninstalled and installed net thrust.

Supersonic Point	TFTJ	MTF	DBE
Uninstalled Net Thrust (K N)	82.5	82.5	81.8
SFC. / Fuel Flow Uninstalled	42.26/3.486	44.51/ 3.672	43.03/ 3.52
Nacelle Drag (kN)	9.771	7.773	10.960
SFC. Installed	47.37	48.79	48.92
Increase In SFC Due To Installation (%)	11.8	9.4	13.4
Subsonic Point			
Required Net Thrust (K N)	56.75	56.75	56.75
Uninstalled Net Thrust (K N)	59	59	57.5
SFC. / Fuel Flow	25.23/ 1.4969	24.58/ 1.4516	26.79/ 1.533
Nacelle Drag(K N)	7.001	6.07	7.055
SFC. Installed	27.72	26.62	29.54
Increase In SFC Due To Installation (%)	11.9	10.3	12.3

Table 5.5 SFCs And Fuel Flows For Uninstalled And Installed Net Thrust

The fuel flow at supersonic off-design performance was assumed to be increased by the same amount that the fuel flow was increased at the supersonic design point in order to compensate the nacelle drag. These increases are 11.9 %, 9.4 % and 13.4 % for TFTJ, MTF and DBE respectively, relative to the supersonic uninstalled fuel flows. In the same way the fuel flows at subsonic off-design performance were increased by 11.9 %, 10.3 % and 12.3 % for TFTJ, MTF and DBE respectively, relative to the uninstalled subsonic fuel flows.

5.5.1 Fuel Bill

In order to restore the net thrust to the required value, the engine mass flow and consequently the fuel flow should increase, relative to the uninstalled values, in order to compensate the nacelle drag. The fuel flow is increased by the same amount the uninstalled thrust is increased. The fuel bill is recalculated for the installed SFCs and fuel flows of the three engines for both mission profiles. Table 5.6 shows the uninstalled and installed fuel bills for both missions.

	MISSION I			MISSION II		
Uninstalled	**TFTJ**	**MTF**	**DBE**	**TFTJ**	**MTF**	**DBE**
SFC(sup/sub)	42.26/25.23	44.51/24.58	43.03/26.79	42.26/25.23	44.51/24.58	43.03/26.79
Fuel Flow kg	3.486/1.497	3.672/1.452	3.52/1.533	3.486/1.497	3.672/1.452	3.52/1.533
Fuel Bill	176777	178178	180833	166756	172345	167420
Installed						
SFC(sup/sub)	47.36/27.72	48.79/26.62	48.92/29.54	47.36/27.72	48.79/26.62	48.92/29.54
Fuel Flow kg	3.908/1.635	4.025/1.570	4.001/1.699	3.908/1.635	4.025/1.570	4.001/1.699
Fuel Bill kg	189148	188469	195167	178111	181799	180779
Difference%	7 %	5.8 %	7.9 %	6.8 %	5.5 %	8 %

Table 5.6 Uninstalled And Installed Fuel Bill.

5.6 Effect Of Bypass Ratio On Supersonic Drag

The above results showed the installed performance when the engine is sized in order to meet the take-off noise regulations. It is important to compare these results with the installed performance when the engine is sized to give the best supersonic performance. The supersonic bypass ratio will determine the inlet LP compressor inlet diameter, therefore, this parameter will be the key issue in determining the size of the engine and the nacelle. Due to its simplicity, the TFTJ engine is chosen to study the above effects.

First, the SFC and the specific thrust are calculated for a fully expanded convergent-divergent nozzle when the bypass ratio varies from 0.2 to 1.2 at the supersonic mode (TET =1800 K, OPR =7.04 and an intake pressure recovery of 0.93). Then, the nozzle exit area is reduced in order to be equal to the LP compressor inlet area, the nozzle is under expanded, and the new SFCs and specific thrust are recalculated using Turbomatch program. Finally, the sizing of the intake, engine, and the nozzle is evaluated for each bypass ratio and the nacelle drag is estimated. The final installed performance for the different bypass ratios and engine diameters are obtained.

		Full Expansion			Under Expansion			Installed	
BPR/ LPC Diameter		SFC	Specific Thrust.	Nozzle exit Area	SFC	Specific Thrust.	Nozzle exit Area	SFC	Specific Thrust.
1.2	1.68	40.38	257	4.85	46.86	221	2.60	52.35	198
1	1.60	40.50	282	4.45	45.66	250	2.40	50.46	226
0.8	1.52	40.73	312	4.07	44.63	285	2.20	48.82	261
0.6	1.44	41.11	348	3.70	44.70	320	2.00	48.53	295
0.4	1.36	41.68	391	3.34	45.02	363	1.80	48.51	337
0.2	1.27	42.65	447	2.97	45.60	418	1.60	48.79	391

Table 5.7 Effect Of Under Expansion And Installation on SFC and Specific Thrust.

Table 5.7 shows these results. Figure 5.8a and 5.8b show the effect of the nozzle under expansion and the installation on the SFC and the specific thrust for different supersonic bypass ratios.

Figure 5.8a shows that the SFC reaches its minimum value, when the nozzle is under expanded, for a bypass ratio of around 0.7, which was selected as the design value. However, considering the installation effects, the minimum SFC occurs when the bypass ratio is around 0.5. This curve is a very powerful tool to determine the best supersonic bypass ratio which gives the best overall performance.

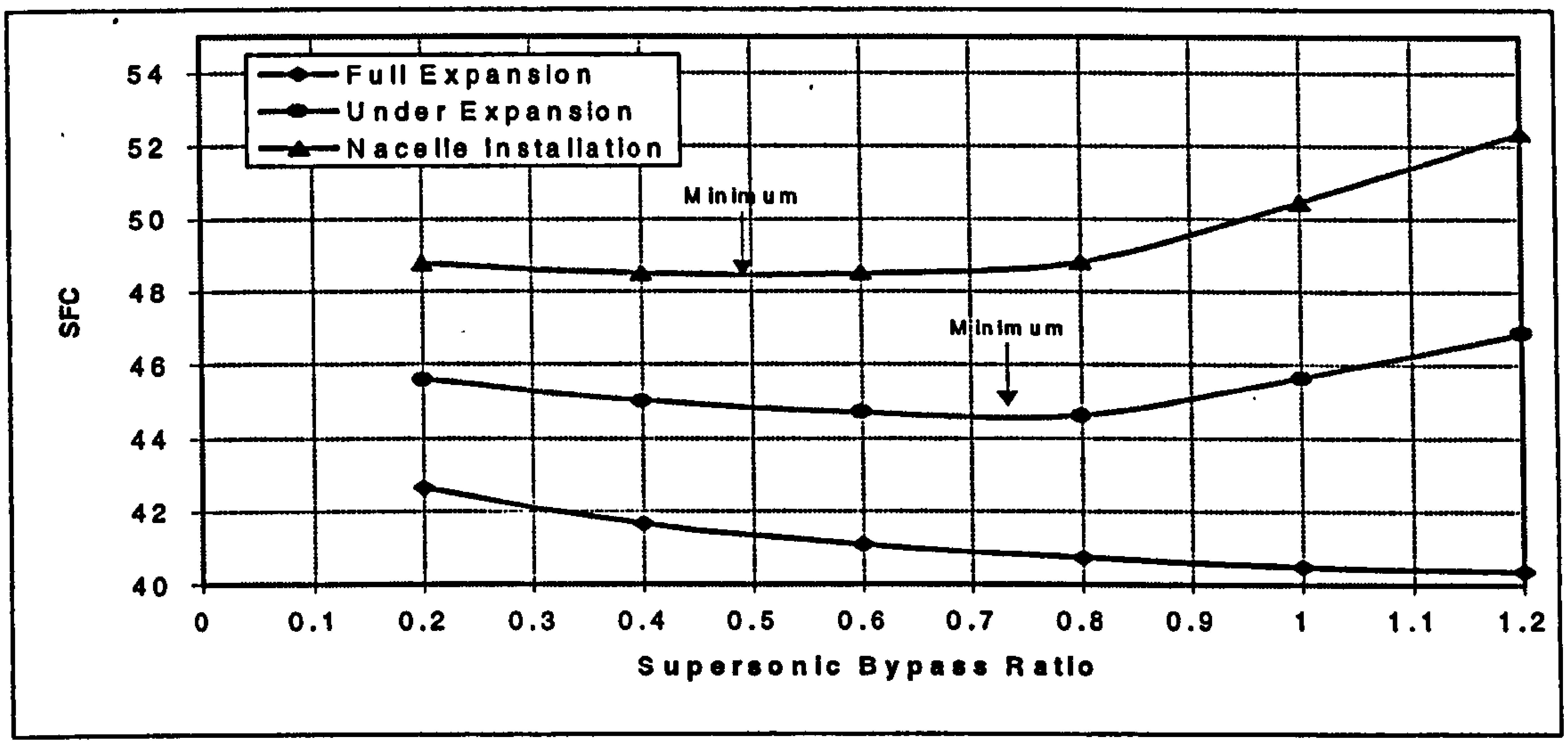


Figure 5.8a Effect Of Nozzle Under Expansion And Installation On SFC.

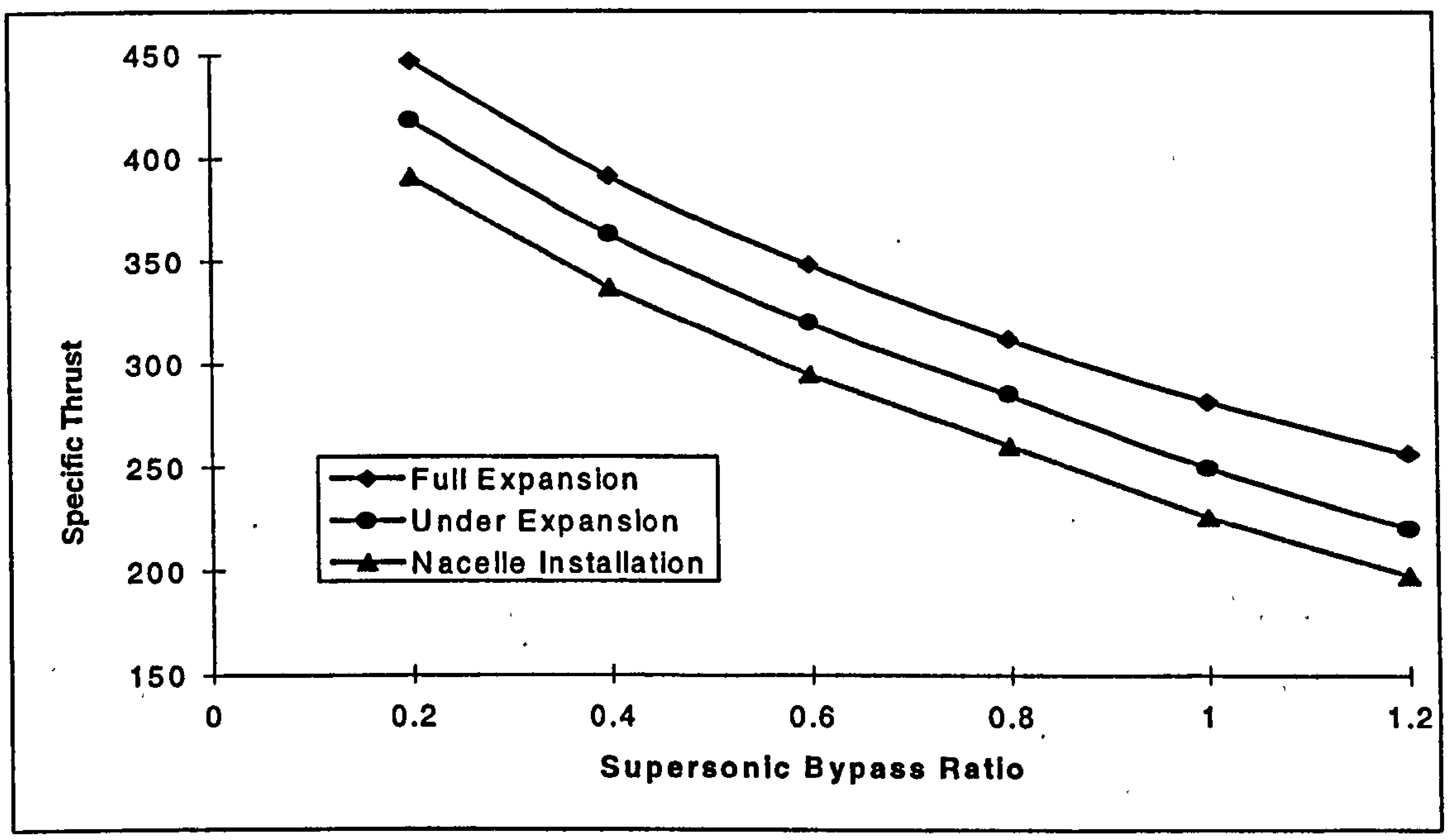


Figure 5.8b Effect Of Nozzle Under Expansion And Installation On Specific Thrust.

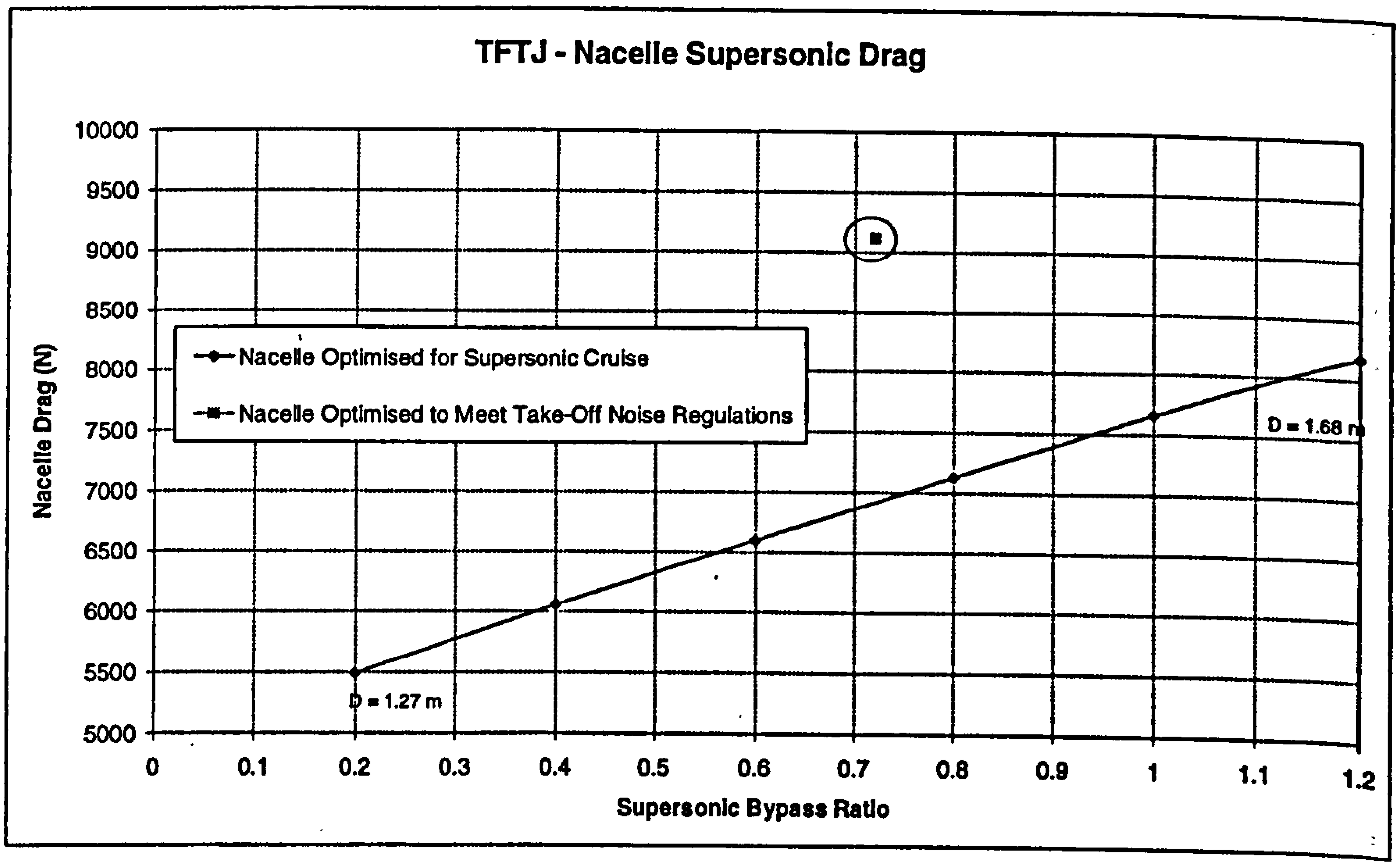


Figure 5.9 Effect Of Nacelle Diameter On Drag.

Figure 5.9 shows the nacelle supersonic drag when the engine is optimised for supersonic cruise and when it is optimised in order to meet the take-off noise regulations. At take-off the engine should have a large diameter in order to pass the relatively high mass flow to keep the exit velocity low. Its drag at supersonic cruise is around 32% higher than if it was designed in order to give the best supersonic performance. This penalty is not acceptable, and other solutions should be found in order to overcome this problem. Using auxiliary intakes or an ejector should be analysed in further works.

Figure 5.9 shows that increasing the engine diameter by 32% would increase the nacelle drag by around 50 %. This indicates the importance of maintaining the engine diameter as low as possible in order to minimise the supersonic drag.

5.7 Remarks And Conclusion

1- From table 5.4 we can see that the skin friction and form drag is the most important term in the total nacelle drag at supersonic mode. It forms almost the total nacelle drag for the MTF. In fact, if the intake inlet area is bigger or equal the nacelle maximum area, the wave and pre-entry drags are negligible. The wave drag is maximum for the DBE where the intake inlet area is smaller than the nacelle maximum area (Fig. 5.4).

2- At subsonic mode, the pre-entry drag is higher and it depends on the intake inlet area and the engine required massflow. Table 5.4 shows that the pre-entry drag is 24 %, 27 % and 24 % of the total nacelle drag for the TFTJ, MTF and DBE respectively. In fact, this shows the necessity of using a variable intake inlet area. If the intake has a fixed inlet area (the same as at supersonic point) the pre-entry drag will be unacceptable at subsonic mode. Figure 5.3 shows the variation of the free stream captured area relative to the supersonic design area. The area required at subsonic mode is approximately half the supersonic area.

3- We can see from Table 5.2 that the intake length is about half the total nacelle length. Decreasing the intake length would decrease the intake total pressure recovery and consequently the net thrust. A compromise should be made between the intake length and the total pressure recovery.

4- The estimated total nacelle drags are as follows:

	Supersonic			**Subsonic**		
	TFTJ	MTF	DBE	TFTJ	MTF	DBE
Thrust Required		82500 N			56750 N	
Total nacelle drag N	9771	7772	10960	7007	6070	7056
Percentage %	8.5%	7.4%	8.6%	12.3%	10.7%	12.4%

The lowest nacelle drag is the MTF engine and this is due to its lowest diameter. Other kinds of drags, such as the interference drag, could influence the nacelle total drag. This kind of drag needs a wind tunnel test in order to be evaluated, which is outside the scope of this work.

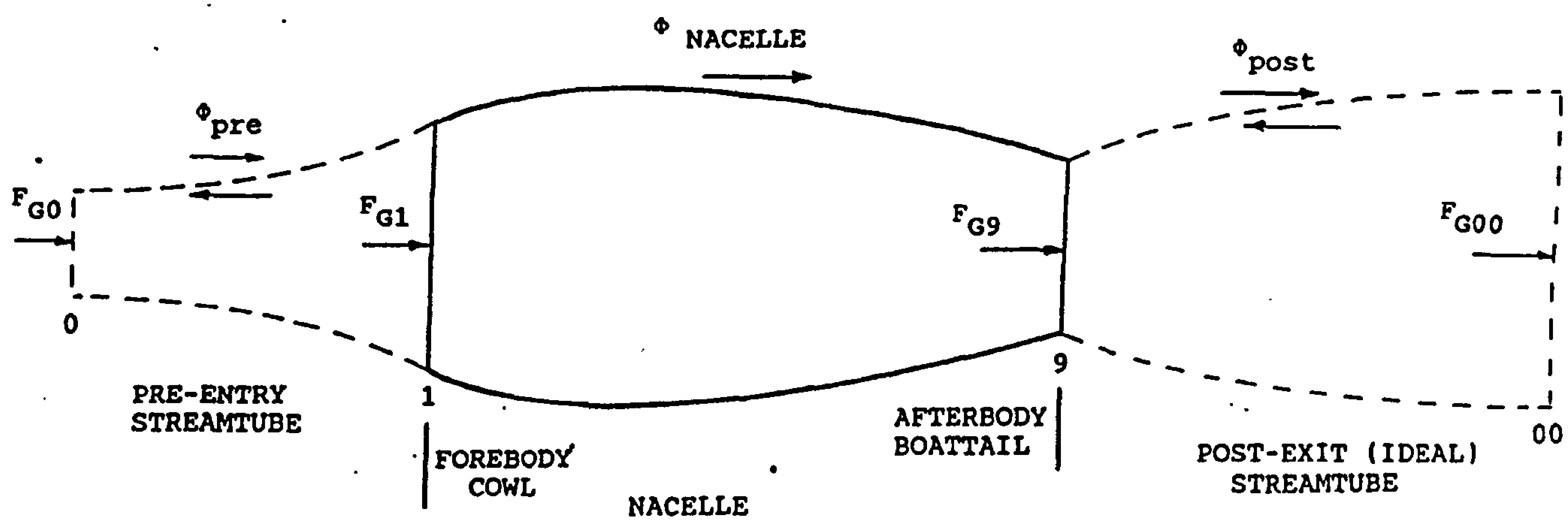
5- Table 5.6 shows the fuel bill for the installed engines for the two missions. We can see, for the first mission between London and Los-Angeles (Supersonic part and Subsonic part), that the best engine is the MTF, but it is the worst for the second mission (only supersonic between Los-Angeles and Sydney). That means that the MTF is not the most adapted to supersonic flight, but is best when the mission is a mix of supersonic and subsonic flights.

6- Figure 5.8 shows that the best supersonic bypass ratio for uninstalled performance is around 0.72. But when taking installation into consideration the best supersonic bypass ratio is around 0.5. However, the difference in SFC, between the two bypass ratios, is very small on the installed performance.

7- Oversizing the engine in order to meet the take-off noise consideration will lead to a large increase in nacelle supersonic drag. This solution is unacceptable in practice and a genuine solution should be found. The use of ejector should be considered.

8- Both TFTJ and MTF are potential candidates for the future SST. For the uninstalled performance the TFTJ appeared to give the best results, but for the installed performance, the MTF became the one which gave the best results. However, the difference in fuel bill between the two engines, for both missions, is very small, less than 1 % . Further studies in different areas are required in order to obtain more information about these engines.

FIGURE 5.1 Forces Acting On A Single-Stream Nacelle (Idealized)



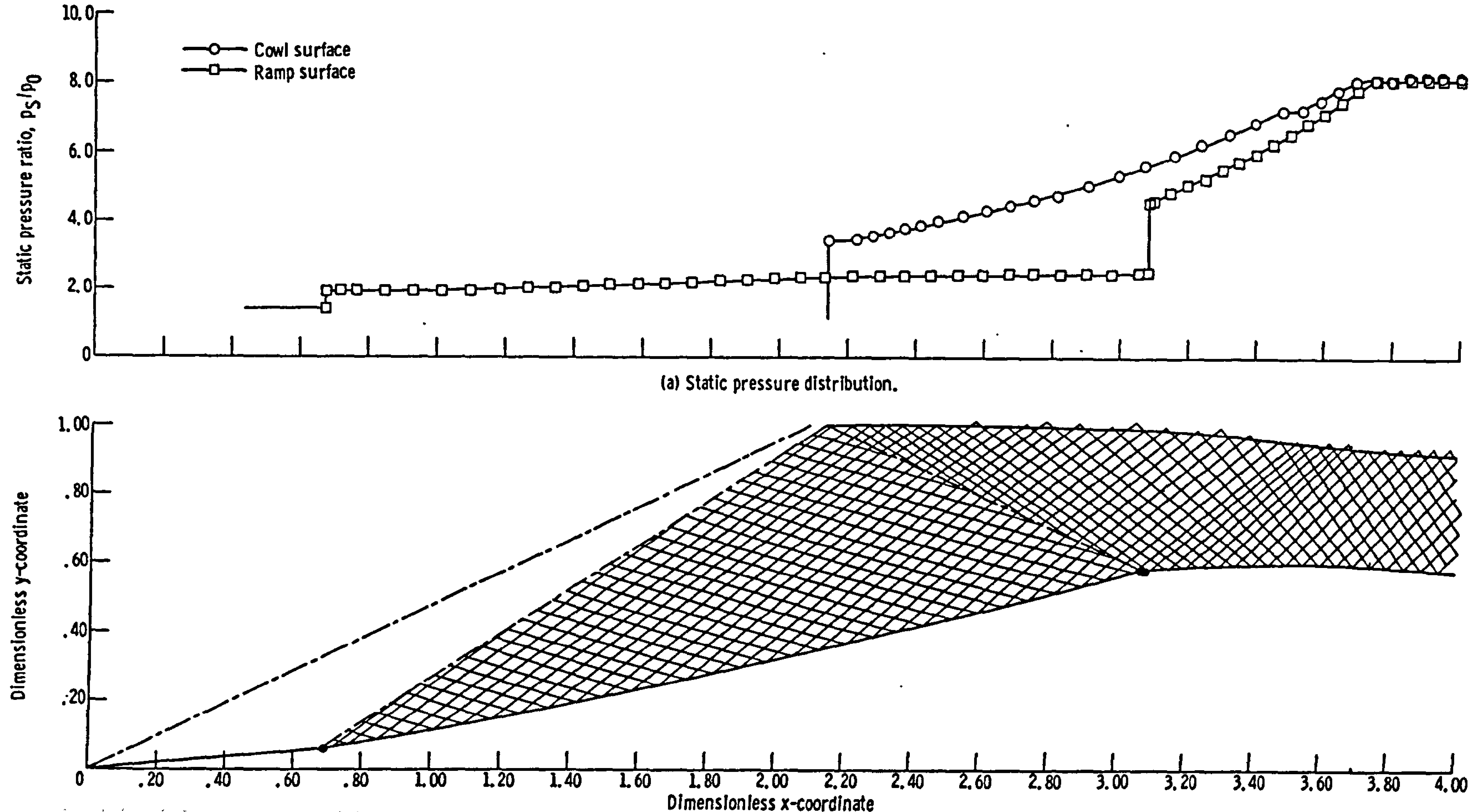
$F_{G0} = \frac{W\,V_0}{g}$ = INTAKE MOMENTUM DRAG

F_{G00} = IDEAL FULLY-EXPANDED GROSS THRUST

F_G ~ STREAM GROSS THRUSTS

Φ ~ NACELLE AND STREAMTUBE FORCES

FIGURE 5.2 Mixed Compression intake Layout



(a) Static pressure distribution.

FIGURE 5.3 Variation Of Free Stream Captured Area In Function Of Flight Mach Number

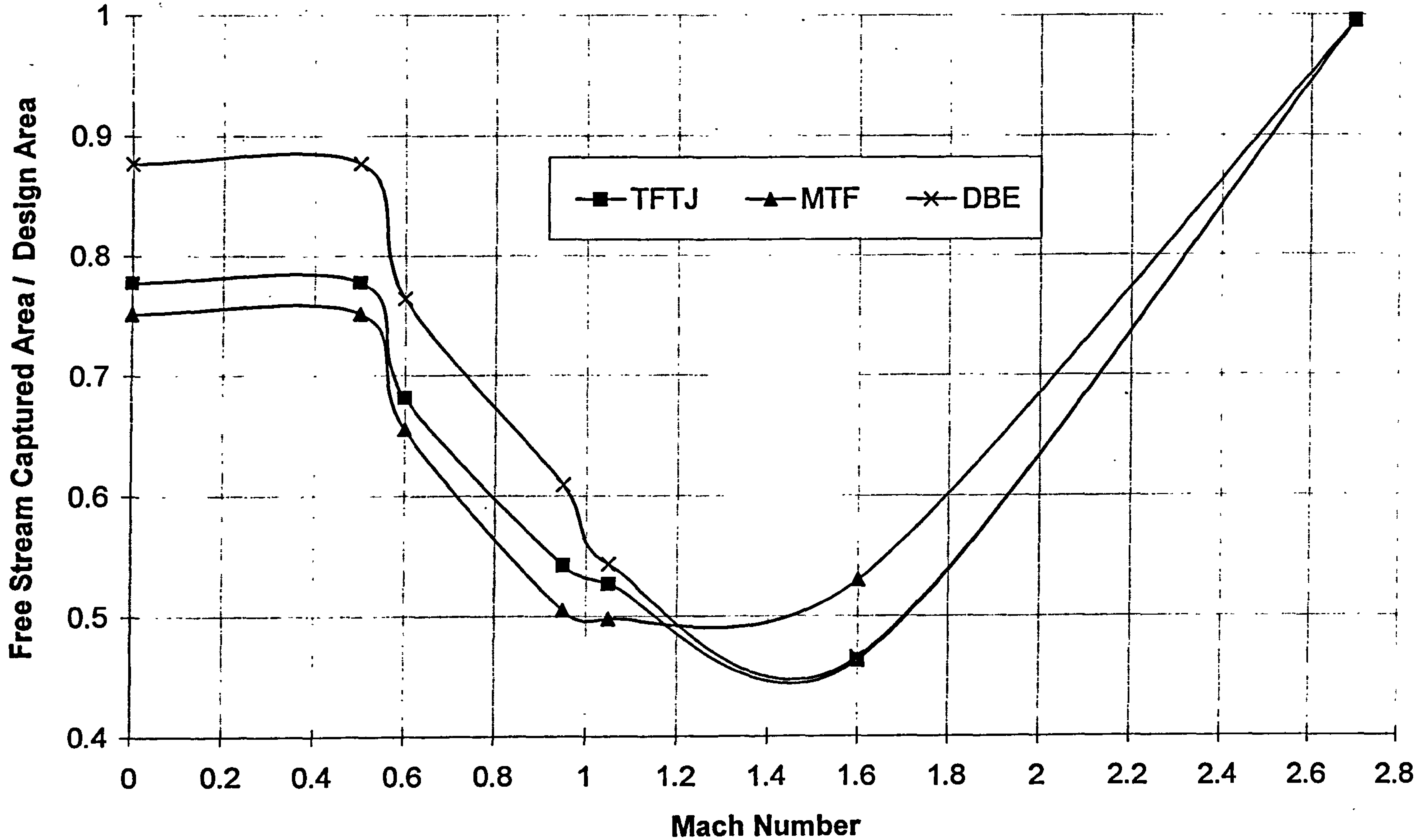
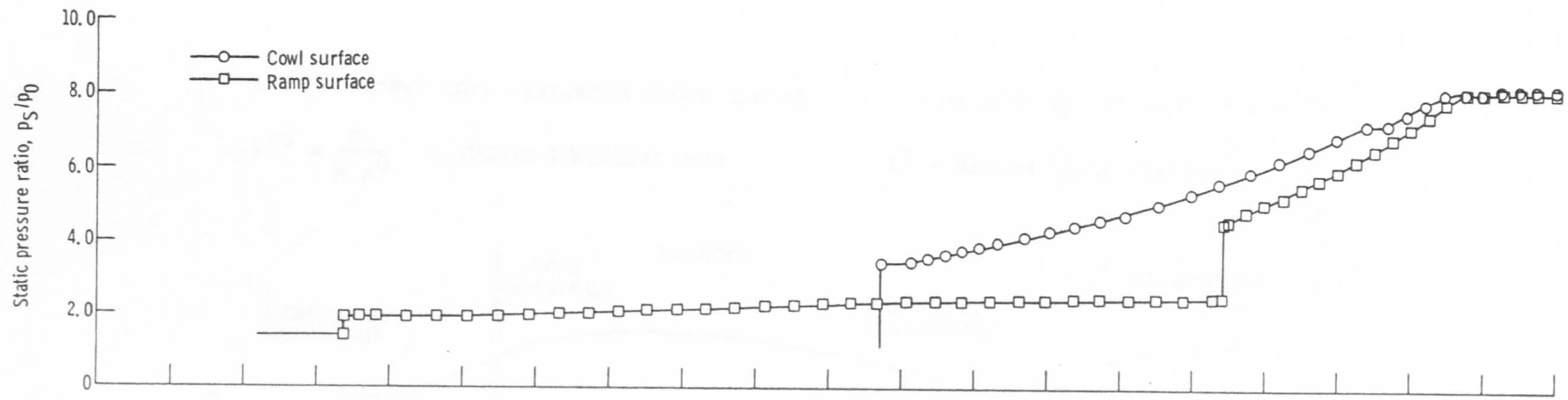


FIGURE 5.2 Mixed Compression intake Layout



(a) Static pressure distribution.

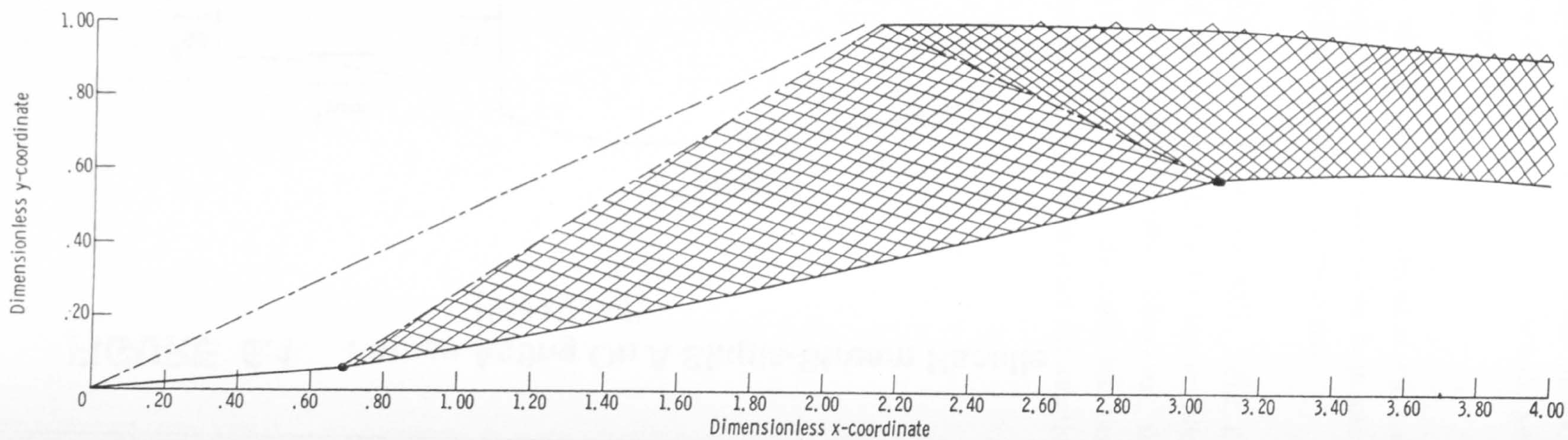


FIGURE 5.3 Variation Of Free Stream Captured Area In Function Of Flight Mach Number

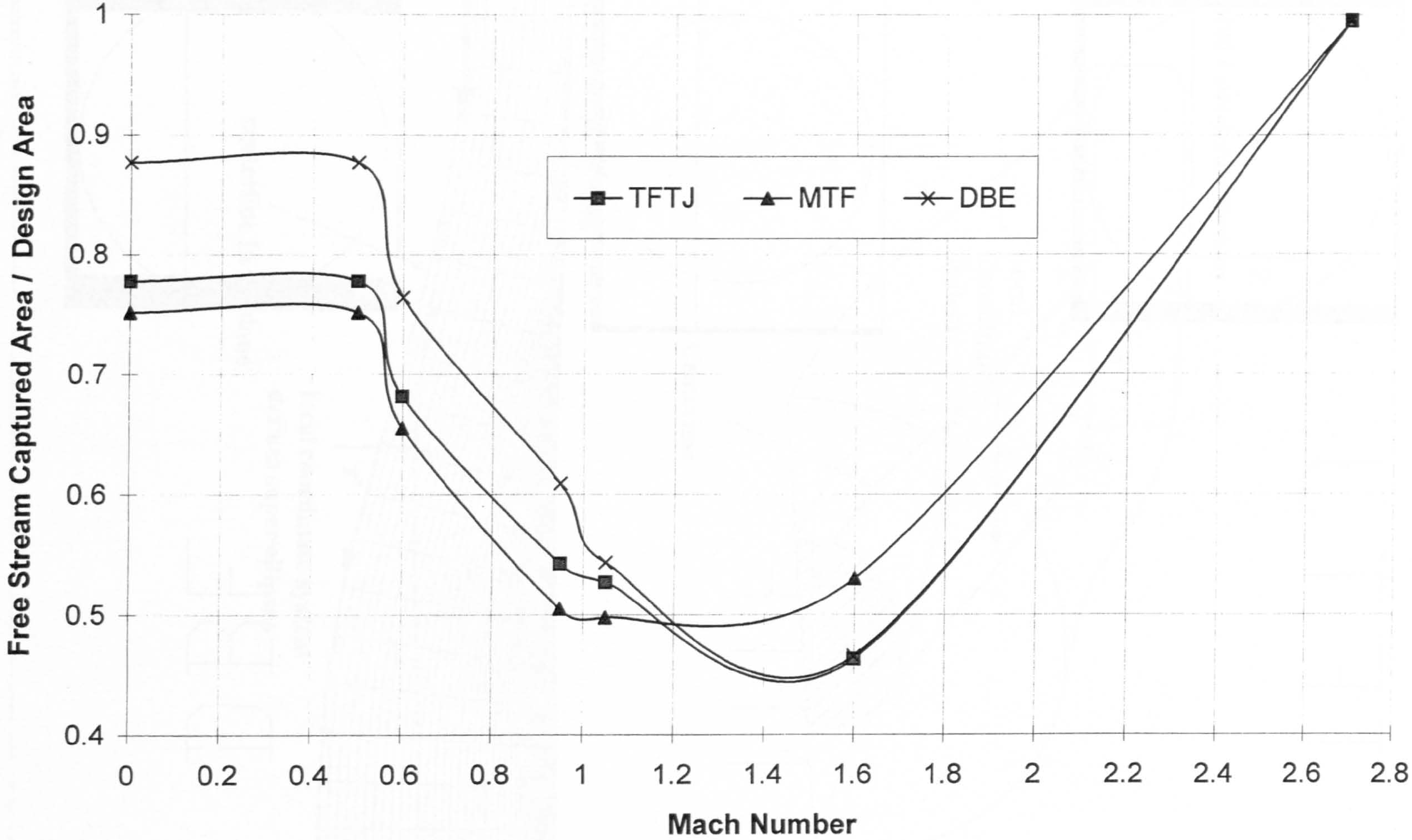


FIGURE 5.4 **Layout Of The L.P. Compressor And The Intake Inlet Area**

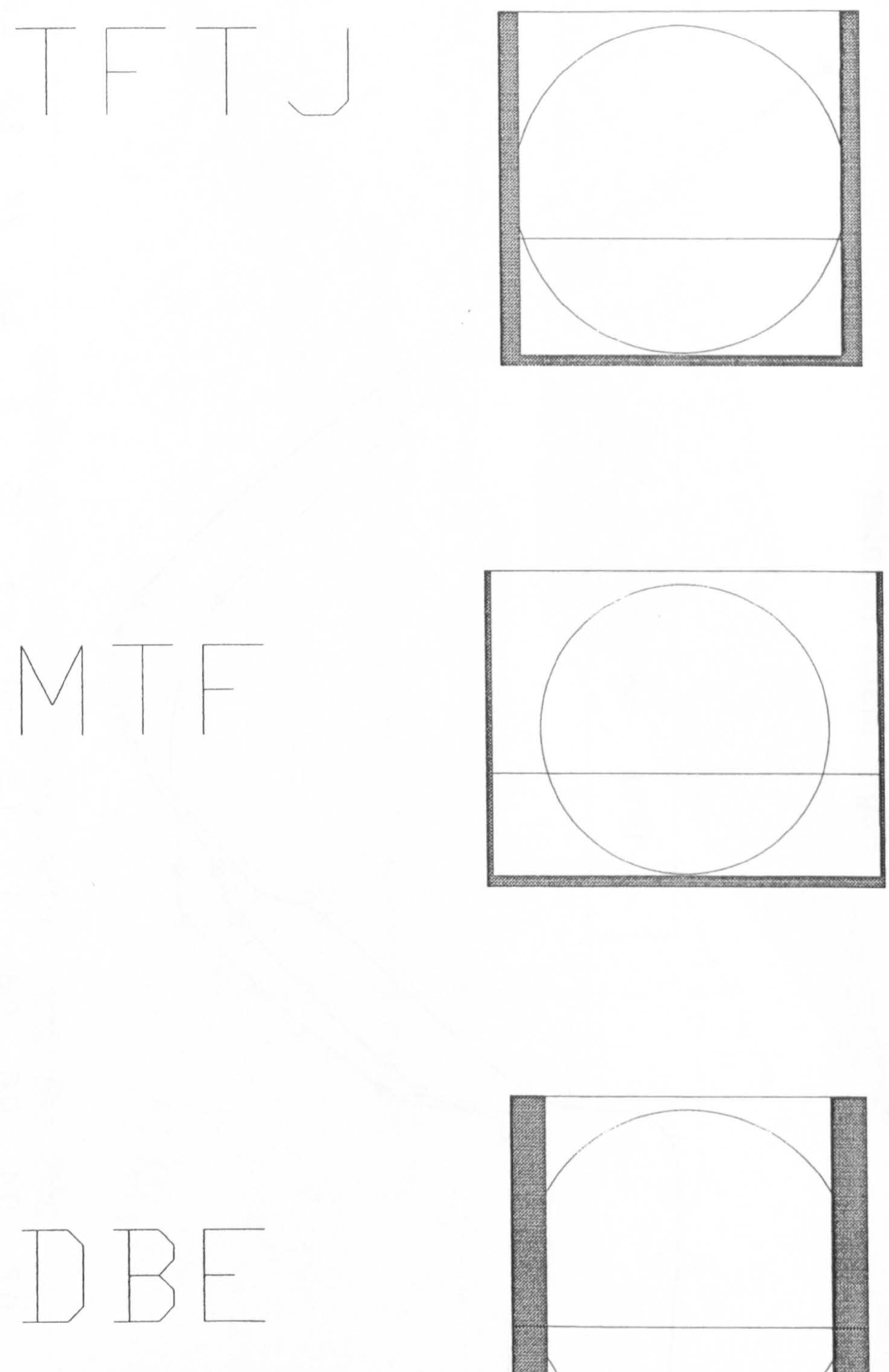


FIGURE 5.5 Intake Subsonic Diffuser

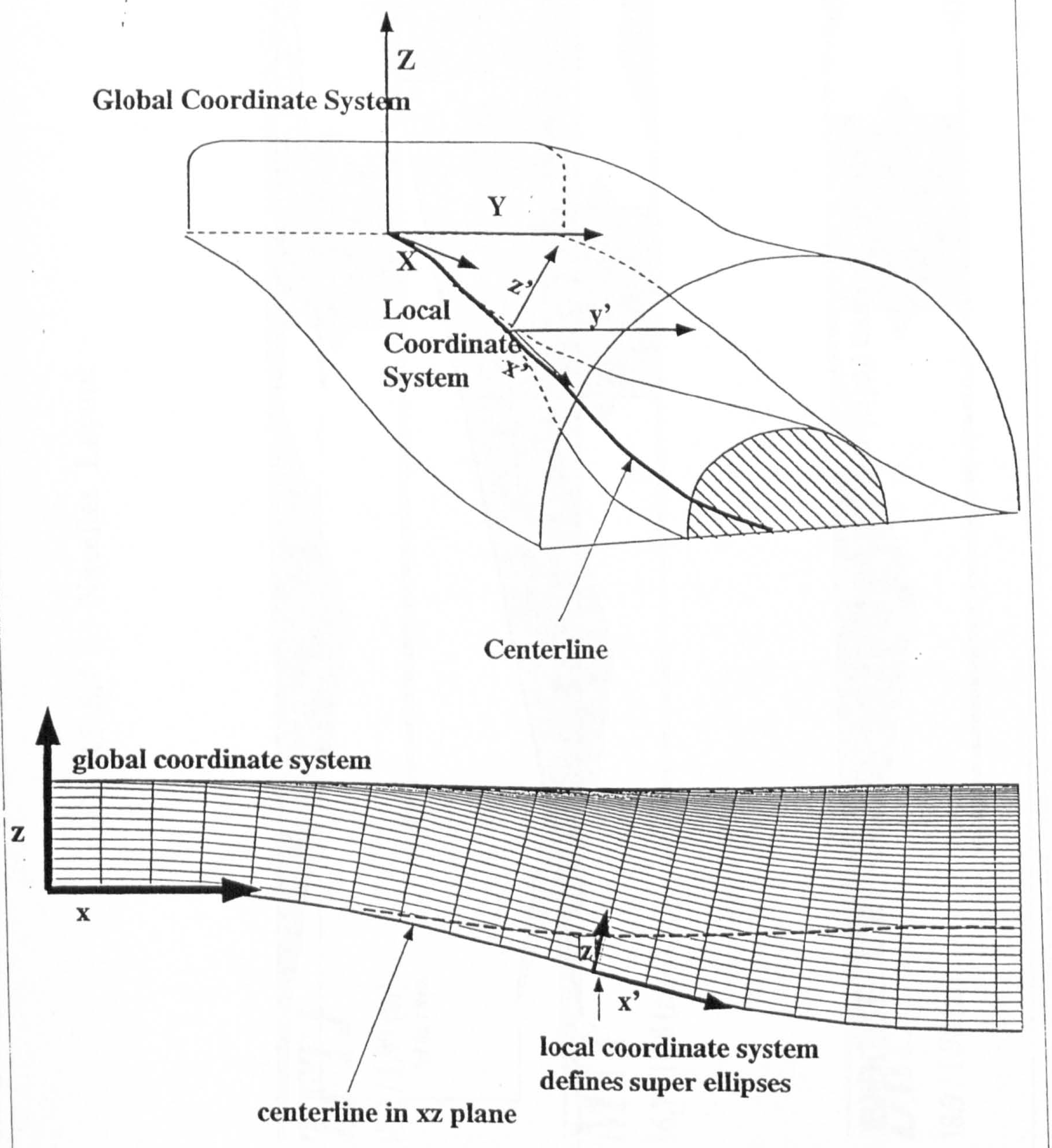


FIGURE 5.6 Variable Geometry Intake Inlet area

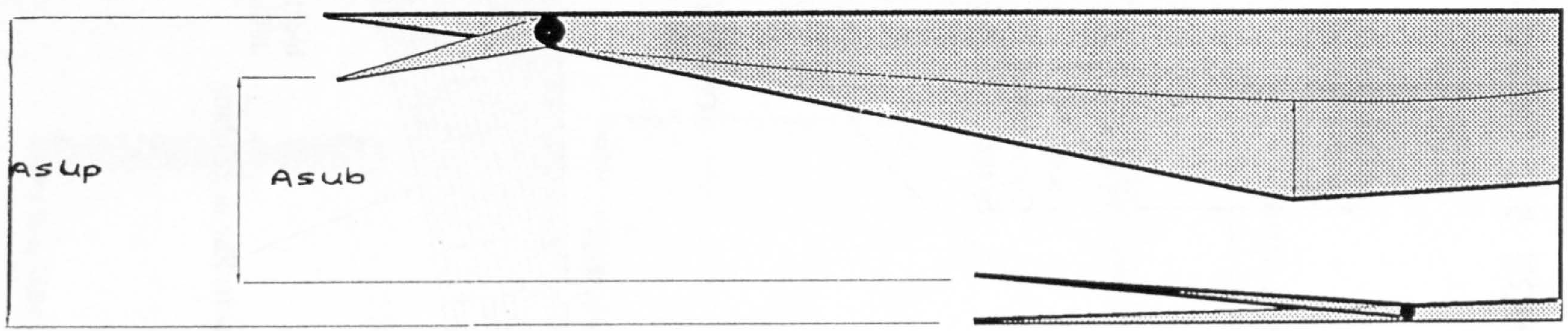
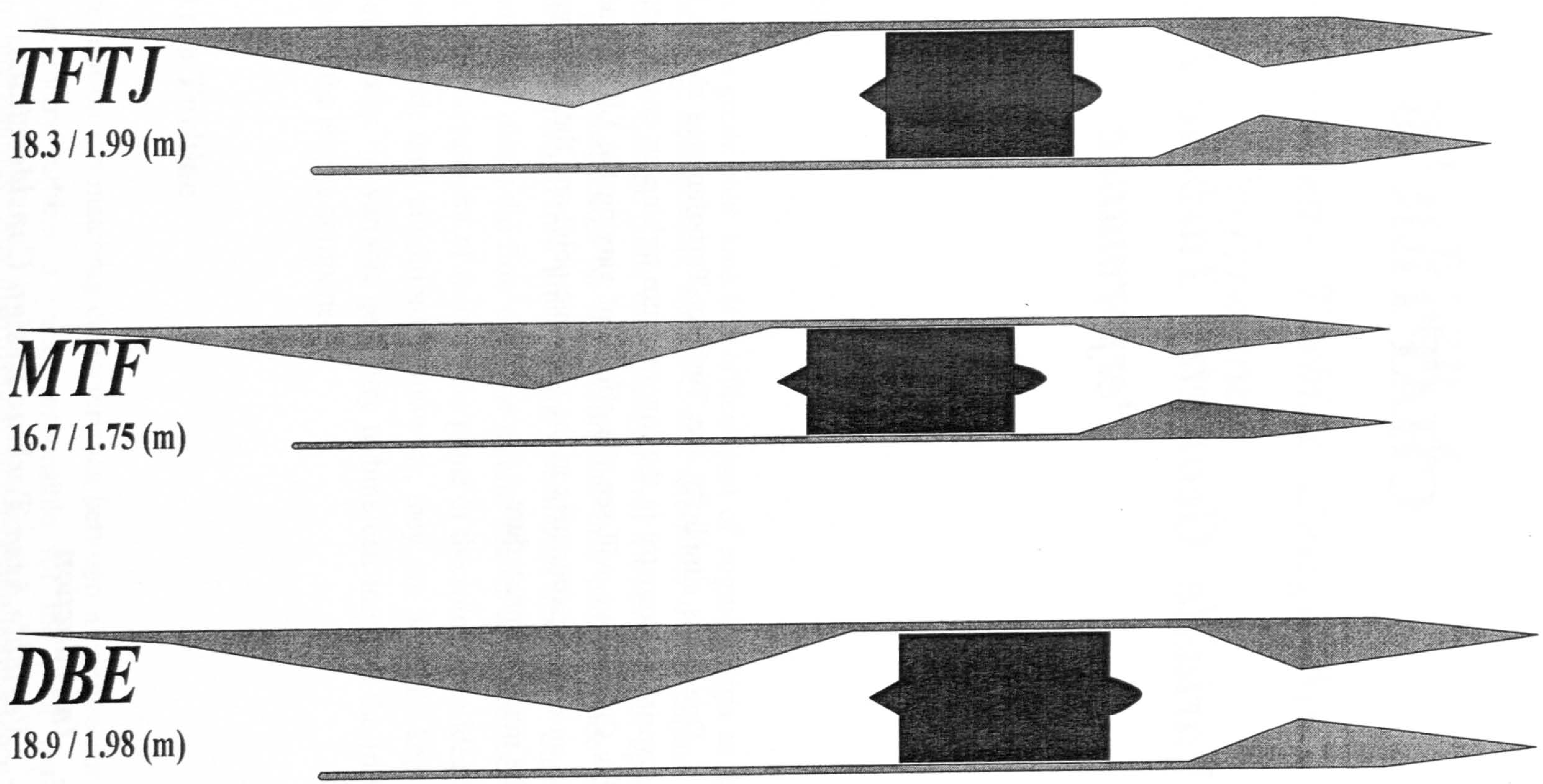


FIGURE 5.7 Nacelles Layout



CHAPTER 6

Variable Geometry Turbine Design And Performance

Due to its simplicity, the Turbojet-Turbofan was chosen to study the effect of using variable geometry in the low pressure turbine on engine performance and design. In this Chapter we will see the effects of varying the LP turbine guide vanes stagger angle on engine parameters at the subsonic point and the consequence of these changes on the matching procedure.

CHAPTER 6

Variable Geometry Turbine Design And Performance

6.1 Introduction

The turbine nozzle guide vane area is a critical part of engine design as its size greatly impacts the surging, acceleration, and SFC characteristics of the engine. The turbine is normally designed with the guide vanes choked over a wide range of operating conditions and this puts a restriction on the excursions that can be made on the compressor characteristics which in turn limits engine operational flexibility. Since one duty of the guide vanes is to direct the flow onto the turbine blades at the correct angle in order to produce a large component of force in the plane of the rotor, then ideally, the vane angles should vary with flow conditions. However, they are normally kept fixed due to mechanical complexity. A variable geometry turbine can improve the matching characteristics of some of the engine components.

6.1.1 Variable Area Turbine

The restriction put on the matching characteristics between a compressor and a turbine by a fixed geometry turbine is partly responsible for the poor part load performance of gas turbine engines. Any improvement in off-design performance would require some degree of flexibility within the engine, but the specified mode of operation will determine whether this flexibility should be provided by the compressor or turbine or both.

Since variable geometry in compressors is deployed passively, it can mainly affect the internal performance of the compressor in question. On the other hand, variable geometry in turbines affects the entire engine cycle as it is active in nature, so therefore, there is some scope for cycle improvement at off-design by the use of such turbines.

The three most important variables which determine the off-design characteristics of any gas turbine are:

- Air Massflow
- Turbine Entry Temperature (TET).
- Overall Pressure Ratio (OPR).

Airflow and turbine inlet temperature primarily determine the output of the engine whilst TET and overall pressure ratio dictate the level of cycle efficiency and hence, SFC. Component efficiency affect both output and SFC but these are of less importance. The components of any gas turbine engine match such that a specified combination of these three variables gives a unique combination of output and SFC, but the type of engine or the control method may impose a restriction on the possible combination of these three variables at off-design.

Conventional engines with fixed geometry turbines must reduce fuel flow and hence, turbine entry temperature, to reduce power. With shaft power cycles, SFC rises with decreasing power whereas with jet engine cycles the beneficial effects of increased propulsive efficiency as thrust decreases may or may not outweigh the detrimental effects of reduced thermal efficiency with decreased thrust. Therefore, one cannot conclude whether or not SFC increases with reduction in thrust.

In general it can be proved (Ref. 11) that as turbine entry temperature decreases, compressor pressure ratio decreases and the effect of this on mass flow outweighs that of a decrease of TET to give a decrease of mass flow as temperature is reduced to decrease thrust.

The reduction in cycle pressure ratio will reduce the thermal efficiency whereas the propulsive efficiency will increase, but to a lower level, as mass flow decreases, compared to that which would have been obtained from a higher mass flow. Also, the reduction in mass flow will cause an increase in both the spillage and boattail drags associated with the inlet and nozzle respectively, as the airflow demanded by the engine is much lower than which the inlet can handle, and the nozzle area will have to close down to accommodate the reduced mass flow thereby providing more aft-end projected area. The reduction in both airflow and pressure ratio causes the compressor to operate over a wide range on its characteristics with poor efficiency at some points. Therefore, it is important to hold both airflow and pressure ratio at high levels while thrust is modulated with temperature.

The variable geometry turbine, within certain limitations, can alter the flow-temperature-pressure ratio matching characteristics of a gas turbine such that any of these variables, or any combination, can be controlled to improve cycle performance. There are other operating advantages of the variable area turbine which make it an attractive element for engine performance control.

6.1.2 Potential Operating Advantages Of Variable Geometry Turbine

Since a change of turbine area affects the engine cycle, there may be a restriction on the amount of area change used as limits on variables such as surge margin, spool speed, etc., on other components may be exceeded. Furthermore, a turbine shares the work in driving the compression system with other turbines in a multi-spool engine, and in jet engines the nozzle is also involved in the pressure ratio split. Therefore, employing a variable area turbine may require one or more of the other components in the expansion system to vary its area as well. Nonetheless, the advantages to be gained with VAT could be enormous.

6.1.2.1 Inlet-Engine Matching

The inlet of an aircraft engine has an airflow characteristic independent of the engine and when the two are installed in the aircraft, they will match efficiently at some operating point whereas a mismatch at other points could lead to excessive drag on the aircraft with poor system performance. This is most critical for high Mach aircraft especially when flying at subsonic speeds where the engine has been throttled back considerably.

With high speed aircraft, the inlet would normally be sized to pass the maximum airflow encountered at altitude, and this is most likely to occur at a high mach number. The nozzle area will be quite large and therefore aft-end drag will be minimal. At the low power settings required for subsonic cruise, the inlet air handling capability exceeds engine demand and therefore some air is spilled around the intake resulting in a large increase in drag. Also, the exhaust nozzle area, if variable, will have to close down due to the lower exhaust flow, giving rise to a large aft-end drag.

The spillage drag can be eliminated if the fan or LP compressor can be controlled with the aid of VAT to swallow all the airflow passed by the inlet, and therefore the aft-end drag is reduced as the exit area is larger to pass a higher mass flow. This problem of inlet-engine mismatch is solved in present day aircraft systems by the use of complicated variable geometry inlets. A variable area turbine coupled with a simpler inlet would be more attractive.

6.1.2.2 Surge Control

Another promising operating advantage of the VAT is that of the surge control. Variable geometry compressors are used extensively in current engines to raise the surge line at low and medium spool speeds but at the expense of a reduction in airflow at a given spool speed. If low speed operation is caused by high mach flight, then the available power is reduced. On the other hand, a VAT can be used to lower the operating line with an accompanying increase in mass flow at fixed rotational speed. The VAT is an equally strong candidate for surge control, as is the variable geometry compressor. Since the deployment of both compressor and turbine variable geometry to control surge could be restricted by limits on some variables. A combination of variable geometries in these two components could be used effectively for surge control such that no limits are encountered.

6.1.2.3 Extended Performance

It may not be possible for a gas turbine engine to deliver more thrust or power in certain operating conditions as a result of one or more components reaching an operating limit. The limits encountered are usually spool speed and TET. A VAT can be used to improve engine performance by increasing one or more variables while keeping the limiting variable at bay.

From the above discussions it can be seen that variable geometry can significantly improve the off-design performance of a gas turbine engines.

6.1.3 Influence Of Turbine Area On Engine Matching

When a gas turbine engine is operating steadily at off-design, the various compatibility equations are maintained and as a result, the components match such that each component operates along a working line. The components at the "hot end" have a controlling influence on the engine cycle and may be used to alter or shift the working line of some of the other components. Therefore, a turbine has an important role to play when determining the performance levels that can be obtained at off-design.

As mentioned earlier, a critical area of turbine design is the turbine nozzle area. Turbines are normally designed to operate choked over most part of their operating rang. This characteristic imposes severe restrictions in the operation of a compressor coupled to the same shaft, and hence, the engine. It is known that for a single spool fixed geometry turbojet, when turbine inlet temperature decreases, the compressor and turbine match such that both compressor mass flow and pressure ratio decrease. This can be taken as generally true for conventional gas turbines. An examination of the effect of the choice of the designed turbine area on engine performance will indicate that any attempt of designing a VCE should include the extensive use of the variable area turbine.

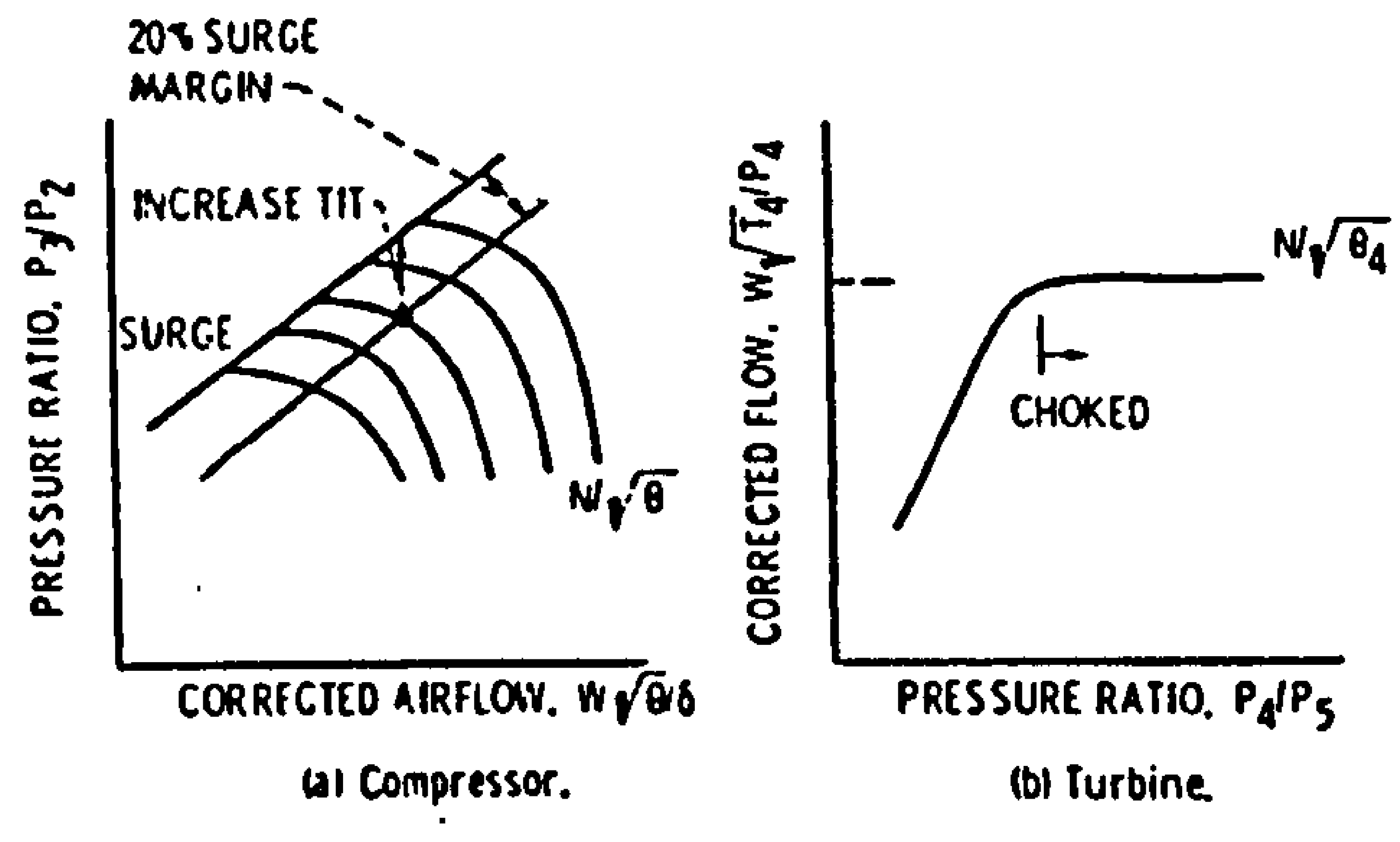


Figure 6.1 Matching Characteristics Of A Single Spool Turbojet.

Typical characteristics of a fixed geometry compressor and turbine for a single spool turbojet are shown in Figure 6.1. The turbine is choked for most of its operating range. In matching the two components, the compressor pressure ratio and mass flow change when turbine inlet temperature changes such that the turbine non-dimensional mass flow remains fixed if the turbine is operating in its choked region; that is:

$$(W.\sqrt{T} / A.P) = \text{Constant}$$

For a prescribed mass flow, W, a given fixed area turbine requires the compressor to operate at a high pressure ratio, that is, closer to surge, for a high TET design, and towards choke for low TET designs. The upper limit on TET is therefore restricted by the amount of surge margin required, materials, and cooling techniques, whereas compressor efficiency and propelling nozzle area tend to restrict the lower bound on TET. Therefore, a given compressor and turbine combination dictate the bounds on TET excursions for an engine.

Since the level of thrust produced by an engine is directly linked to the turbine inlet temperature, a larger turbine nozzle area is indicative of a high value of thrust while a small area gives lower thrust levels for a given compressor design point. The larger turbine area design permits the engine to be operated at the maximum TET over most of the mach number range whereas the smaller turbine area design requires a lower TET in order to prevent the compressor from surging. A larger nozzle area is therefore desirable for high mach flight although at the expense of a high specific fuel consumption.

The disadvantage of a high design nozzle area with its attendant high TET is that the engine has to be severely throttled back to deliver the required cruise thrust at subsonic speeds. The decrease in airflow necessary to achieve this does not only reduce propulsive efficiency but also increases spillage and aft-end drags giving rise to poor installed performance. On the other hand, a smaller nozzle area requires a lesser reduction in TET to deliver the subsonic cruise thrust and therefore incurs lower installation losses.

6.2 Variable Area Turbine Design Considerations

The satisfactory performance of a variable area turbine requires that the turbine be able to handle a wide range of inlet non-dimensional mass flow over a wide range of non-dimensional speed while at the same time maintaining acceptable efficiency levels. The ability of the turbine to accomplish this requires a study of the flow conditions within the turbine for the particular application, and it may be necessary to conservatively design the turbine to allow for satisfactory or acceptable performance at conditions widely removed from the design.

The compromises to be made on turbine design for the particular application may require a design range rather than a design point. The characteristics and limitations of the variable area turbine impose undue design considerations which are absent in the design of fixed geometry turbines.

6.2.1 Variable Area Turbine Design

Turbine flow capacity can be changed in two ways either by restaggering the stator blades or by restricting the annulus area. Whilst the first method, restaggering the stator blades, allows the flow to be increased above nominal, the second method, restricting the annulus area, is usually preferred as being mechanically simpler to introduce.

6.2.1.1 Variable Stagger Blading

The turbine stators are manufactured with a circular root fixings in order that the stagger of the stators may be varied (Figure 6.2). The normal variation in flow capacity is quite small but on some turbines the flow could be varied over a range of 17.5%. This 17.5% variation in flow may require a 26% variation in stator throat area (Ref. 26). This method is preferred because the loss in efficiency is generally smaller than other methods.

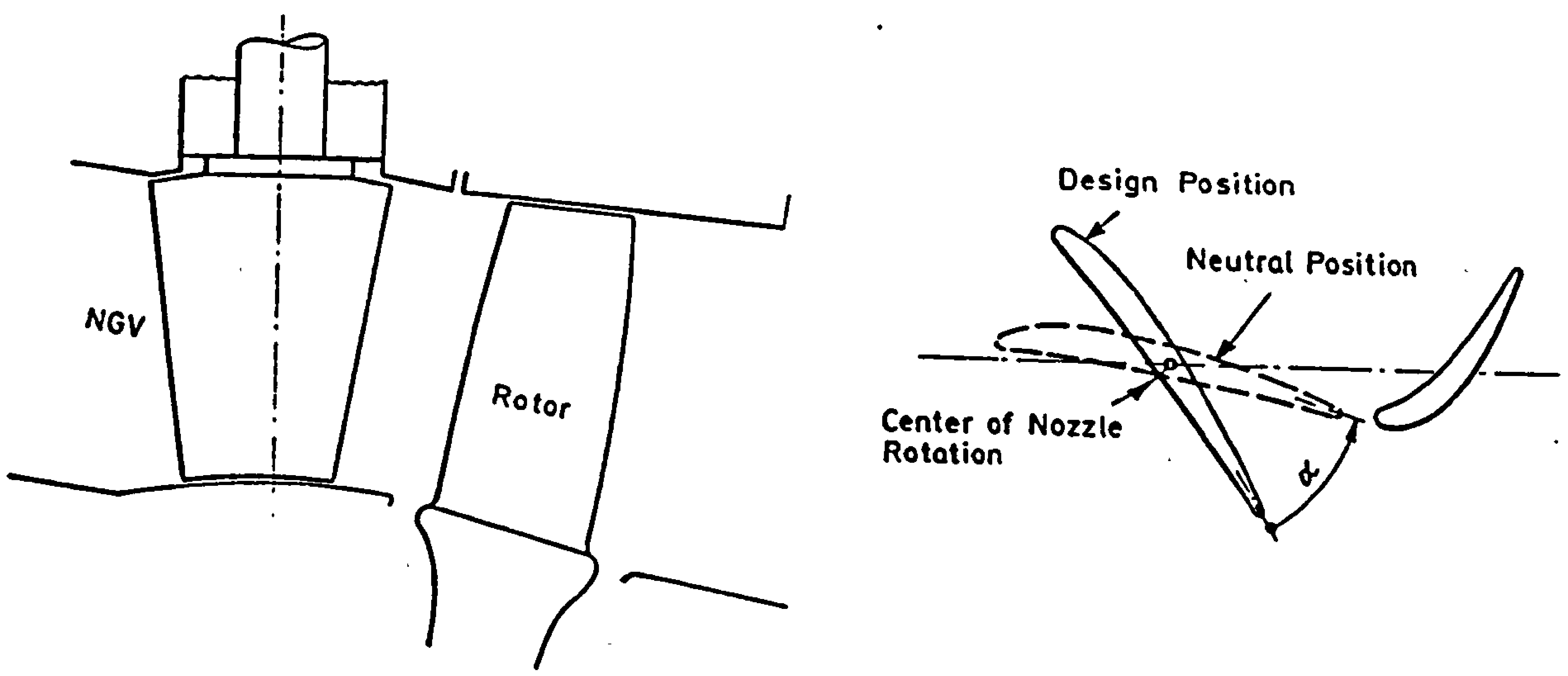


Figure 6.2 Turbine Variable Scheme

6.2.1.2 Variable Annulus Area

There are two obvious ways of restricting the annulus area, either by mechanically introducing an obstruction into the flow or by introducing secondary air, ideally in such a way that the effective flow restriction is bigger than the amount of secondary air. This method is effective and reasonably efficient, but the loss in efficiency is higher than the first method.

6.2.2 Turbine Choking

In a turbine, the entry stationary airfoils, often called nozzle guide vanes or nozzles, expand the entering flow and discharge it into rotating airfoils, known as the rotor or blades. The flow of gas through the turbine nozzles has much in common with the flow of a compressible fluid through an exhaust nozzle, including choking at the minimum area when the back pressure is below the critical value.

Therefore, the flow through the turbine nozzles is a function of the turbine pressure ratio and the turbine inlet total temperature when the nozzles are operating at subcritical pressure ratio and are not choked. Conversely, the flow through the turbine nozzles depends only upon the inlet total pressure and the inlet total temperature, once choking occurs.

The work extraction of modern gas turbines is usually so large, and the turbine exit pressure so high, that the nozzles are usually choked for the design point and surrounding region. If not, the throat Mach number is sufficiently close to one that the non dimensional mass flow approximates that of choking.

The parameters that are normally used to express the performance of a turbine are:

Total pressure ration R_t or Enthalpy drop ratio
Corrected mass flow $(W.\sqrt{T}/P)$
Corrected mechanical speed $(N/\sqrt{T})$
Isentropic efficiency (η_t)

The maximum value of the corrected mass flow is reached at a pressure ratio which produces choking conditions at some point in the turbine. Choking may occur, as is presented before, in the nozzle throats, and either in the rotor blade passages or in the annulus at outlet from the turbine, depending on the design. Choking in the nozzle throats is a more normal situation and then the constant speed lines merge into a single horizontal line as indicated on the mass flow plot of Figure 6.3. Furthermore, for reasons of size, it is desirable that the turbine mass flow rate per annulus area be as large as possible, which points to turbine nozzle choking and a high enthalpy drop ratio.

Even in the unchoked region of operation the separation of $N/\sqrt{T}$ lines is not great, and the larger the number of stages the more nearly the mass flow characteristics can be represented by a single curve independent of $N/\sqrt{T}$. Such an approximation is very convenient when predicting the part load performance of a complete gas turbine unit (off-design calculations).

On the other hand, if choking occurs in the rotor blade passages or outlet annulus, the maximum mass flow will vary slightly with $N/\sqrt{T}$. Then the whole performance of the turbine will be presented by the plot shown in Figure 6.4.

With this kind of operation the corrected mass flow at the turbine nozzle drops quickly as the corrected speed, relative to design, increases to reach the maximum value. After this point, the corrected mass flow increases again until it reaches the value of the choked rotor mass flow rate. This operation has the disadvantage that the off-design performance of the turbine is too difficult to be predicted and thus turbine simulation, with these kind of performance maps, needs more loops and time in order to converge.

6.3 Turbofan-Turbojet Variable geometry Low Pressure Turbine

The turbofan-turbojet engine has two turbines the high pressure and the low pressure one. Employing adjustable guide vanes in the hot section of the engine (HP Turbine) would incorporate a high technological risk, therefore it was decided to use variable geometry only at the low pressure turbine where temperature is lower. The risk still high but less than if it is used at the HP turbine. The guide vanes should only be adjustable and not the rotor blades. Besides this, only a single stage should be considered (Ref. 23).

At the supersonic mode, the TFTJ engine main parameters are defined. These parameters include the TET, overall pressure ratio, bypass ratio, LPC pressure ratio and the HPT non-dimensional mass flow. The selection of the IPC and HPC pressure ratios will determine the LPT non-dimensional mass flow. For one value of the HPT non-dimensional mass flow there is a wide range of the LPT non-dimensional mass flow. This depends on the choice of the IPC and HPC pressure ratios. Table 6.1 shows an example of the variation of LPT NDMF for two extreme choices of these pressure ratios.

TFTJ Engine	Supersonic Mode
BPR	0.72
OPR	7.04
TET	1800 K
HPT NDMF	690

	Extreme Solution 1	Extreme Solution 2
LPC PR	1.68	1.68
IPC PR	1.2 (minimum)	3.493 (maximum)
HPC PR	3.493(maximum)	1.2 (minimum)
LPT NDMF	**1503 (maximum)**	**810 (minimum)**
SFC	41.46	41.38

Table 6.1 Effect Of IPC & HPC Pressure Ratios On LPT NDMF.

Increasing the HPC pressure ratio would increase the LPT NDMF because the HP turbine should deliver more work in order to drive the HP compressor and the HPT exit pressure would drop causing the increase of the LPT NDMF. Figure 6.5 shows the variation of the LPT NDMF in function of the HPC pressure ratios at supersonic and subsonic modes. Figure 6.6 shows the effect of changing the inlet total mass flow (specific thrust) for a bypass ratio of 1.4 on the LPT NDMF.

Figure 6.5 shows that, the gap between the supersonic and subsonic HPC pressure ratios is higher when the subsonic bypass ratio increases. Using variable geometry in the LPT, by closing the NGV, would decrease this gap while maintaining the same bypass ratio.

6.3.1 Effect Of LP Turbine Adjustable Guide Vanes On Engine Performance

Figure 6.7 (Picture below) shows the effect of opening the LPT nozzle guide vanes on the component characteristics. Increasing the LPT choked non-dimensional mass flow (opening the NGV) will reduce the LPT, IPC and LPC pressure ratios while increasing the HPT and HPC pressure ratios. Reducing its choked NDMF would lead to an opposite effect of the above parameters. Figure 6.9 shows the variations of the different component pressure ratios when closing and opening the LPT NGVs from -6° to 10°.

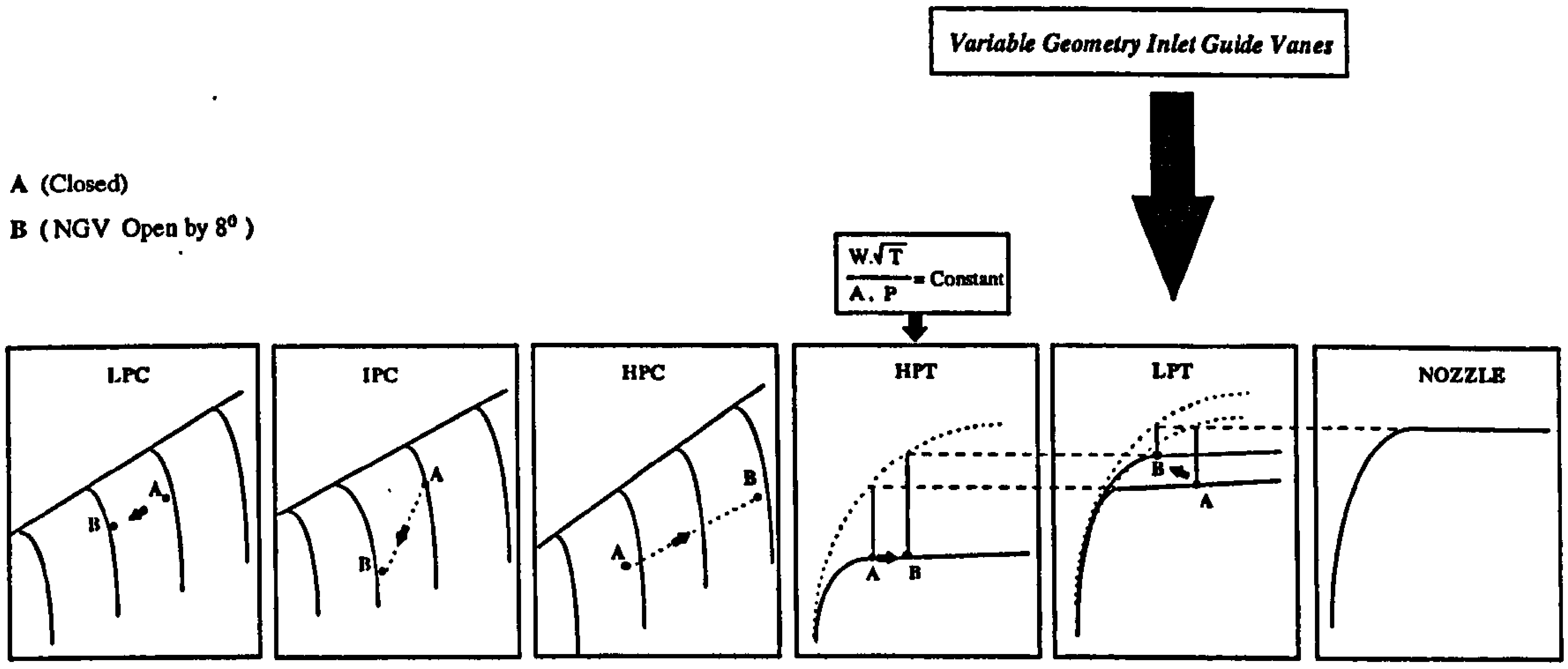


Figure 6.7 Effect Of LPT Variable geometry On Component Characteristics.

Table 6.2 shows the effect of changing the LP turbine guide vanes angle on the Turbofan-Turbojet engine's performance at subsonic mode. All other components geometry kept constant including the nozzle throat, mixing area and the compressor stators, this will show only the effect of changing the LP turbine guide vanes nozzle area on engine parameters. The TET is changed in order to maintain the engine net thrust. These results were obtained by modifying Turbomatch input file, this includes changing the LP turbine NGV angle relative to design point. In practice many components geometry may need to be changed simultaneously in order to remain in a safe zone of operation.

Guide vanes	Closed by 6°	Fixed Geometry	Open by 6°
Total mass flow (kg/s)	255	246	241
TET K	1120	1172	1200
OPR	9	8.7	8.6
BPR	1.39	1.4	1.38
Net thrust (kN)	60.6	60	60.2
SFC	24.26	25.57	26.39
LPT NDMF	1147	1300	1378
LPT (PR/ Efficiency)	3.06/94	2.65/90	2.48/86
HPT NDMF	748	765	771
HPT(PR/ Efficiency)	1.53/87	1.72/90	1.82/90
LPC/ IPC/ HPC PR	1.91/2.43/1.95	1.87/1.84/2.53	1.85/1.63/2.85
Efficiency %	85 / 99 / 90	86 / 90 / 88	86 / 91 / 86

Table 6.2 Effect Of LP Turbine Variable Geometry On Engine Parameters.

6.3.2 Effect On SFC

Table 6.2 shows that SFC decrease by 5.1% whent the NGVs are closed and increase by 3.2% when they are opened. When opening or closing the nozzle guide vane throat area the bypass ratio and the overall pressure ratio (OPR) change in opposite directions. In our case, the bypass ratio and the overall pressure ratio are more or less the same. The differences in SFC come mainly from component efficiency changes. These results reaffirm the results already found in Chapter 3 (Paragraph 3.3). It is worth mentioning that the 5.1% decrease in SFC when the NGVs are closed is due to the 3.4% increase in the overall pressure ratio and better overall component efficiencies.

6.3.3 Effect On Bypass Ratio

The bypass ratio is almost unchanged (Table 6.2). The effect of LP turbine variable geometry on the bypass ratio depends greatly on the HP turbine operating point and TET. When opening or closing the LPT NGVs the OPR changes, hence the core mass flow should change accordingly in order to maitain the HPT NDMF constant. These changes in the core mass flow cause the changes in the bypass ratio.

The HP turbine supposed to be choked during almost the whole mission profile and have the same value of non-dimensional mass flow. However, in our case the HP turbine non-dimensional mass flow is reduced by approximately 2.2% when the NGVs are closed by 6 degrees, and this is caused by the reduction of its pressure ratio beyond the choked value. When opening the NGVs the HPT NDMF remains approximatly constant. This is the reason why bigger changes happen when closing the LPT NGVs than when opening them. Figure 6.8 shows the variations of bypass ratio, overall pressure ratio, SFC and the specific thrust in function of the LPT NGVs angles.

Closing the guide vanes angle has a bigger effect on changing the LP turbine non-dimensional mass flow than opening them. Table 6.2 shows that closing the guide vanes by 6° would decrease the non-dimensional mass flow by 11.8% and an increase of 5.8% when opening them by the same amount.

6.3.4 Effect On Compressor And Turbine Pressure Ratios

The main effect of using variable geometry in the LP turbine is the redistribution of compressor pressure ratios. The overall pressure ratio is more or less constant because it depends directly on the HP turbine matching at this point which has a fixed geometry. Opening the LP turbine guide vanes would increase the HPC pressure ratio and reduce the LPC and IPC pressure ratios in order to maintain a constant overall pressure ratio. Closing them would cause the opposite effects. The effect on the LPC pressure ratio is less significant than on the IPC and HPC pressure ratios.

Increasing the guide vanes throat area would increase the LP turbine non-dimensional mass flow and this increase would be achieved mainly by decreasing the LP turbine inlet total pressure, (the mass flow and overall pressure ratio cannot be significantly changed because the HP turbine has a fixed geometry). This decrease in total pressure would happen if the HP turbine had to yield more work to drive the HPC and this can be done by increasing the HPC pressure ratio. It is for this reason that the HPC pressure ratio increases when the guide vanes area is open. Figure 6.9 shows the variation of the three compressors and the two turbines pressure ratios when opening and closing the LPT NGVs.

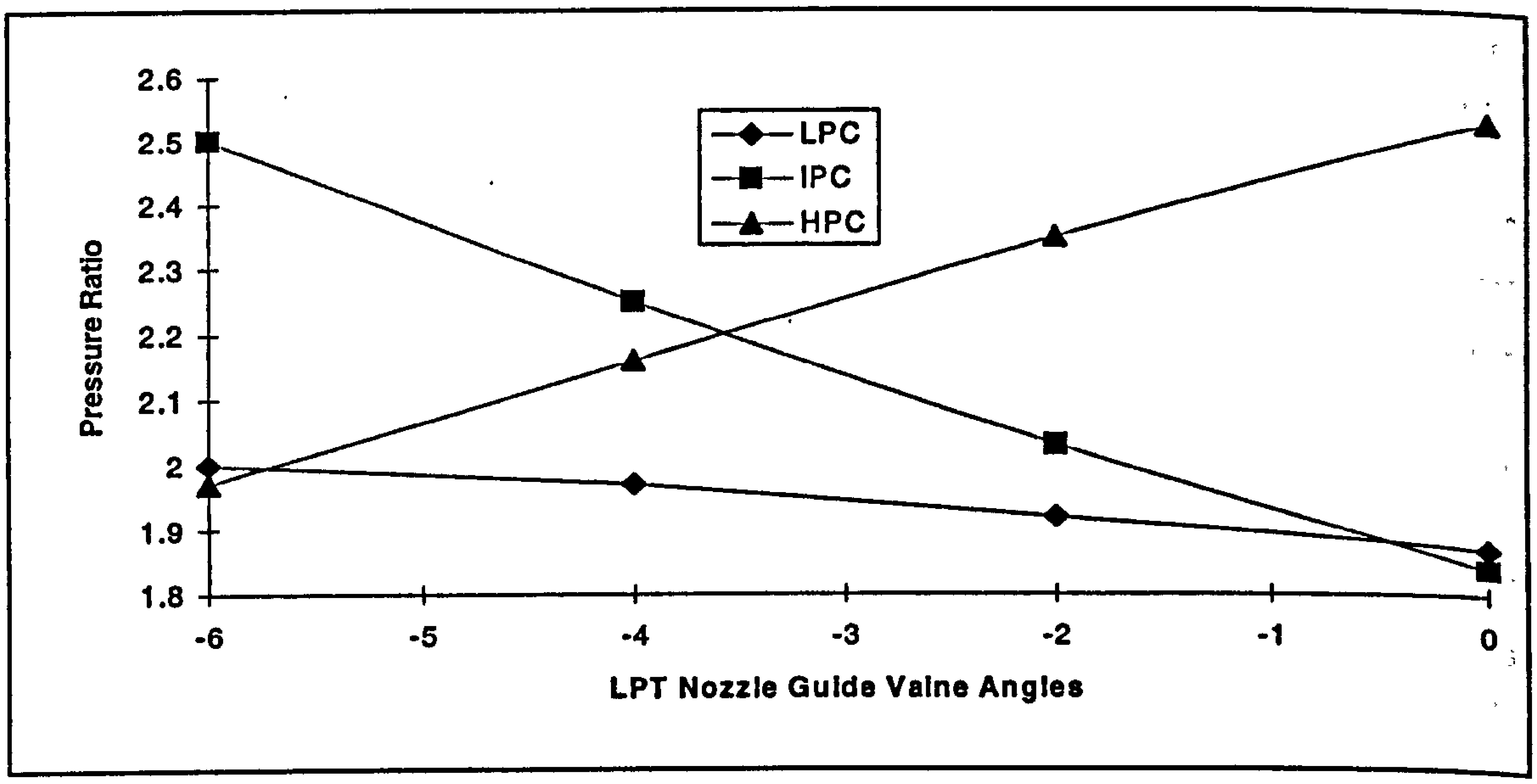
In Chapter 3 (Paragraph 3.3, TFTJ engine), it was found that the lowest SFC, at low specific thrust, is obtained for a bypass ratio of 1.4 Increasing or decreasing the bypass ratio is coupled with reduction or increase in the overall pressure ratio, these changes offset each other and the SFC remain almost the same. Therefor, using variable geometry in the LPT in order to change the bypass ratio will not offer any benefit in terms of fuel consumption.

For this case, Mach 2.7 and 0.95 and for the TFTJ engine, the use of variable geometry in the LPT seemes to have little effect on SFC but a much significant effect on the LPC, IPC and HPC pressure ratios. Therefore, using variable geometry in the LPT could play an important role in the compressor matching between the cycles of a VCE.

6.3.5 Effect On Supercharging The HP Compressor

In the previous paragraphs, while closing or opening the LPT NGVs the TET was changed in order to maintain the required net thrust. It is worth trying to study the effect of using variable geometry in the LP turbine on SFC and net thrust while maintaining every thing else constant. Figure 6.10 shows the changes, relative to fixed geometry, of SFC, BPR, OPR and the net thrust when the LPT NGVs are closed up to 6 degrees.

The most important result is the 15% increase in the net thrust. This increase is due only to supercharging of the HP compressor, while the TET remains constant. Closing the LPT NGVs will decrease the HPC pressure ratio while the LPC and IPC pressure ratios will increase in order to keep the HPT NDMF more or less constant. Consequently, the HPC total inlet pressure is increased. In order to maintain the HPC NDMF constant the core mass flow should be increased to compensate the increase in the total inlet pressure. This increase in core mass flow will reduce the bypass ratio and increase the net thrust. This phenomena is call "supercharging".



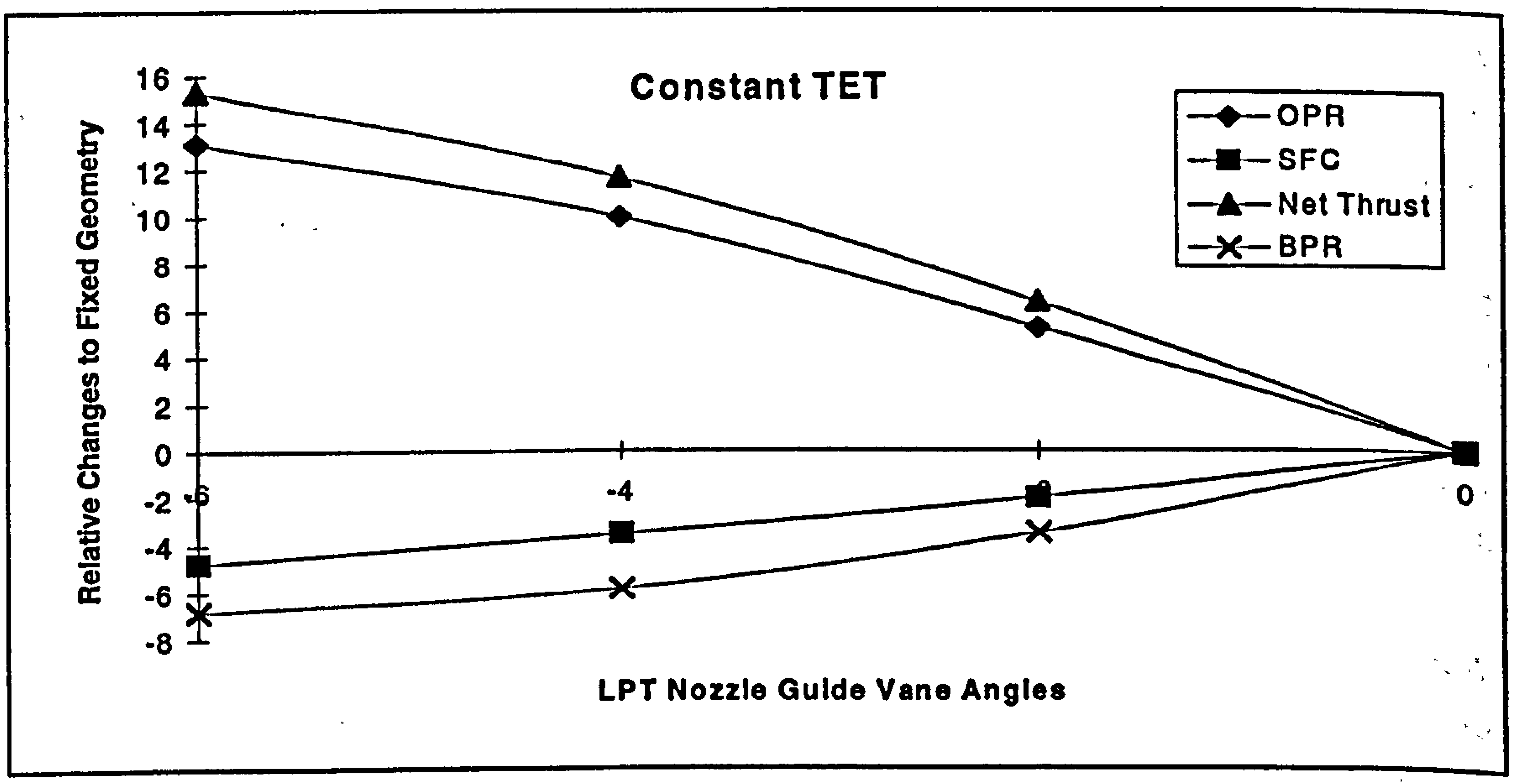


Figure 6.10 Effect Of LPT Variable Geometry On HPC Supercharging.

This phenomenon is interesting where the net thrust can be increased without increasing the TET. This can be used at take-off or at climb and acceleration where extra thrust is needed. This will increase the engine life considerably. This could have some application in military aircraft where thrust can be increased without fuel penalty.

6.4 Effect Of Adjustable Guide Vanes On Turbine Efficiency And Non-Dimensional Mass Flow

In order to estimate the potential benefit of using adjustable guide vanes at the low pressure turbine on the engine performance, it was necessary to evaluate the changes in efficiency and NDMF while closing or opening the guide vanes at the first stage of the LP Turbine. It was decided to use a variable stagger blading in order to change the guide vanes throat area, where the guide vanes are manufactured with a circular root fixing in order to change the stagger of the blades.

The LP Turbine was designed first at the subsonic mode using Turbgeo, Turbdes and Turboff software available at the PPA department of Cranfield University. The turbine maps for different rotational speeds and fixed geometry were obtained using the Turboff program. Initially this program was not designed to study the effect of variable blade stagger on its performance, but in its input file it requires the blade stagger and the blade opening parameters Figure 6.11. Once the blades are manufactured these two parameters are dependent on each other, a manual approximate calculation was done in order to obtain the blade opening in function of blade stagger Figure 6.12. New turbine maps were obtained for each new blade stagger and blade opening values. The stagger blade angle was changed by -5, 5 and 15 degrees relative to design point.

6.4.1 Low Pressure Turbine Design

As previously mentioned, the design point chosen for studying the effect of LP turbine variable geometry on efficiency and non-dimensional mass flow is the subsonic point (Altitude 9150 m, Mach Number 0.95). The LP turbine inlet parameters obtained from the results of Turbomatch program are :

Inlet mass flow:	103 kg/s
Total entry temperature:	957 K
Total inlet pressure:	2.44 Atm
Total temperature drop:	184.5 K
Fuel-Air ratio:	0.01479

The design starts by using the Turbgeo program which provides a rapid determination of design velocity diagrams and calculates annulus geometry as well as all mean aerodynamic parameters. It also generates the input file for the second program

,Turbdes. This program allows for three dimensional effects and estimates design point efficiency. The turbine off-design performance was carried by Turboff which is a pitch line axial flow turbine off-design performance prediction method.

Further information about turbine design can be found in Chapter 4 and more detailed information about the above software can be found in Reference 5. The input and output files are given in Appendix D.

While designing the LP turbine using the previous programs, it has been found that the efficiencies given by three programs at the design point were not the same and the difference was not acceptable. A more detailed examination of Turbdes output file has shown that an error message was given near the end of the file and it indicated that some parameters (space to chord ratio) were out of the scope that the program was designed for. When these parameters were corrected in the initial input file the efficiencies given by the three programs were almost identical.

6.4.2 Effect On Efficiency

Figure 6.13 shows the variations of efficiency in function of blade stagger for the design rotational speed. The design efficiency is 89.4 %, changing the stagger blade angle by -5, 5, 15 degrees relative to design would drop the efficiency to 87.2 %, 87.1% and 83 % respectively. Closing or opening the guide vanes by 5 degrees would drop the efficiency by approximately 2.5 %, opening them by 15 degrees would reduce the turbine efficiency by 7.2 %. These results are in the range expected for this kind of application (Ref. 26).

The changes in efficiency in the LP turbine obtained by using the Turbomatch program are higher than the above results. Table 6.2 shows that opening the guide vanes by 6° would drop the effeciency by 4.4%. Closing them by the same amount would increase the efficiency again by 4.4%.

6.4.3- Effect on Non-dimensional Mass Flow

Figure 6.14 shows the effect of variable geometry on the Non-Dimensional Mass Flow (NDMF). When opening the guide vanes by 5 and 15 degrees relative to design, the NDMF would increase by 7.8 % and 11.7 % respectively, closing them by 5 degrees would reduce the NDMF by 15.6 %.

The changes in NDMF obtained by using Turbomatch are of the same order as the above results. Table 6.2 shows that opening the guide vanes by 6° would increase the LP turbine NDMF by 6 % and a reduction of 12 % when they are closed by 6°.

6.5 Conclusion

Variable geometry has been examined only in the LP turbine. Employing variable geometry in the hot section of the engine (HP turbine) would incorporate a high technological risk. The effective change in the nozzle guide vanes throat area will be done by using adjustable guide vanes stagger angle.

The effect of using adjustable guide vanes on turbine efficiency is important. An efficiency drop of 2.5% and 7.2% will occur when the guide vanes are open by 5° and 15° (relative to design point) respectively. The increase in non-dimensional mass flow will be 7.8% and 11.7% for the same above angles. When the guide vanes are closed by 5° the efficiency and the NDMF drop by 2.5% and 12% respectively. These results were obtained by using Turbdes and Turboff programs which use relatively old technology, more sophisticated methods could improved the above results.

The effect of using variable geometry in the LP turbine on bypass ratio, SFC and specific thrust at the subsonic mode is very small, the SFC and the bypass ratio remains virtually constant. The major effect of LPT variable geometry is on changing the IPC, HPC and the turbine pressure ratios. The changes in the LPC pressure ratio are less significant. The LPT variable geometry is a very effective tool to influence the component characteristics, its use along with variable geometry compressor increase the chances of better cycles matching for the VCE. This advantage should be weighted carefully against the complexity and the technological risks associated with its use.

LPT variable geometry could be used to change the subsonic bypass ratio and therefore improve the fuel consumption. This improvement depends on the variable cycle engine design and on the supersonic and subsonic Mach numbers and altitudes. In our case increasing the bypass ratio will not lead to a lower SFC (Figure 3.5).

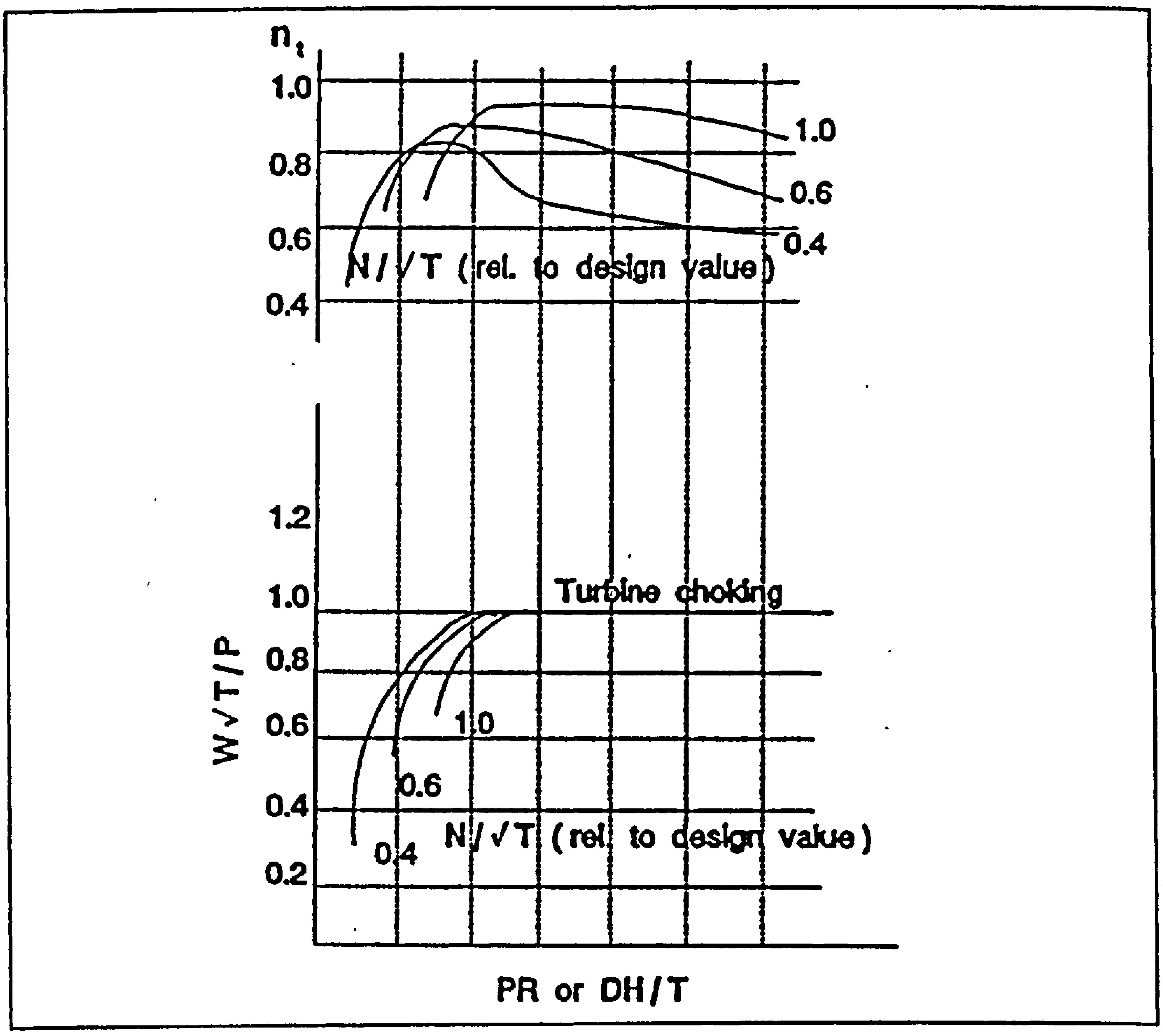


FIGURE 6.3 Turbine Characteristics With The Nozzle Choked.

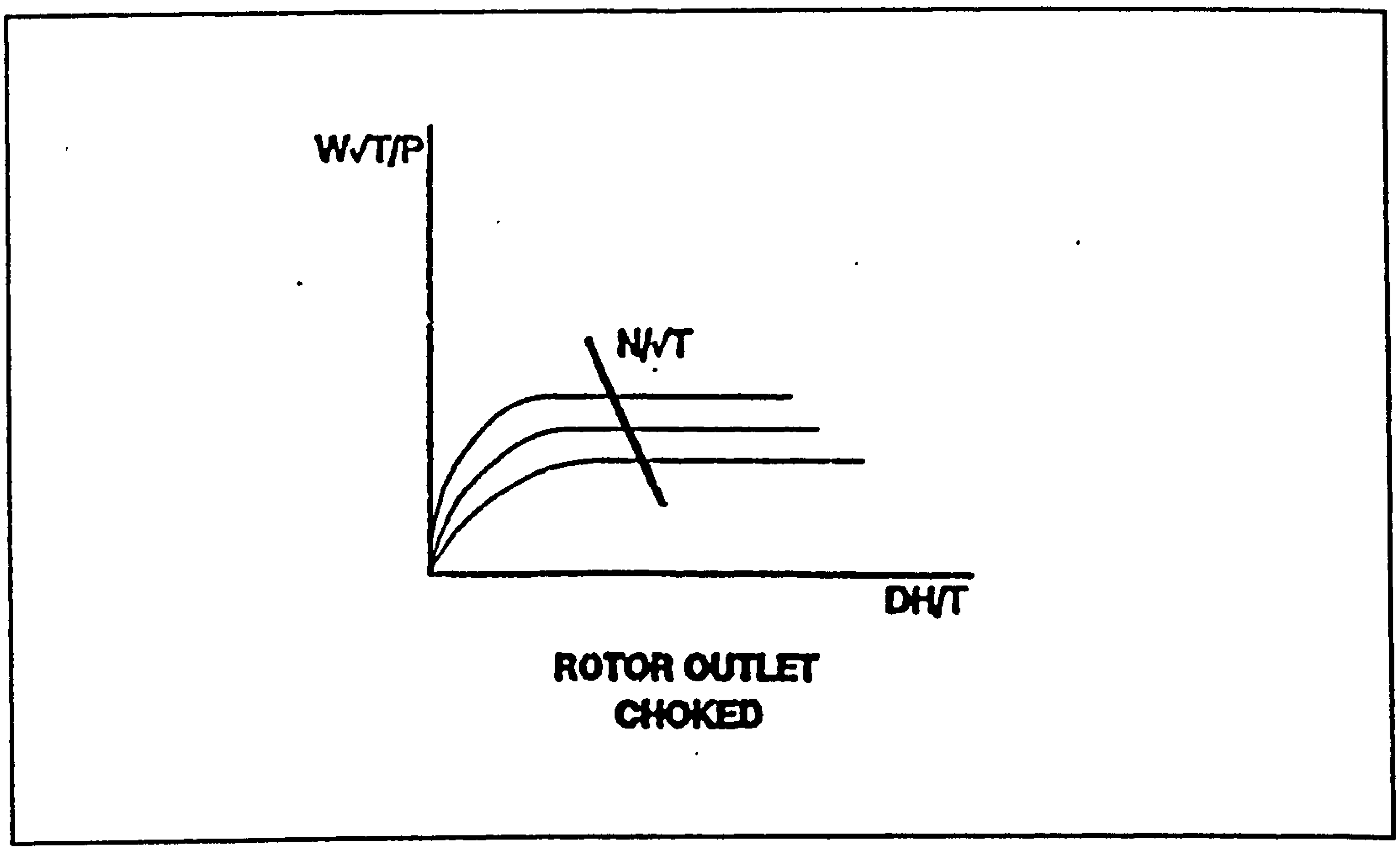


FIGURE 6.4 Turbine Characteristics With The Rotor Choked.

FIGURE 6.5 Effect Of HPC Pressure Ratio On LPT Non-Dimensional Mass Flow For Different Overall Pressure Ratios

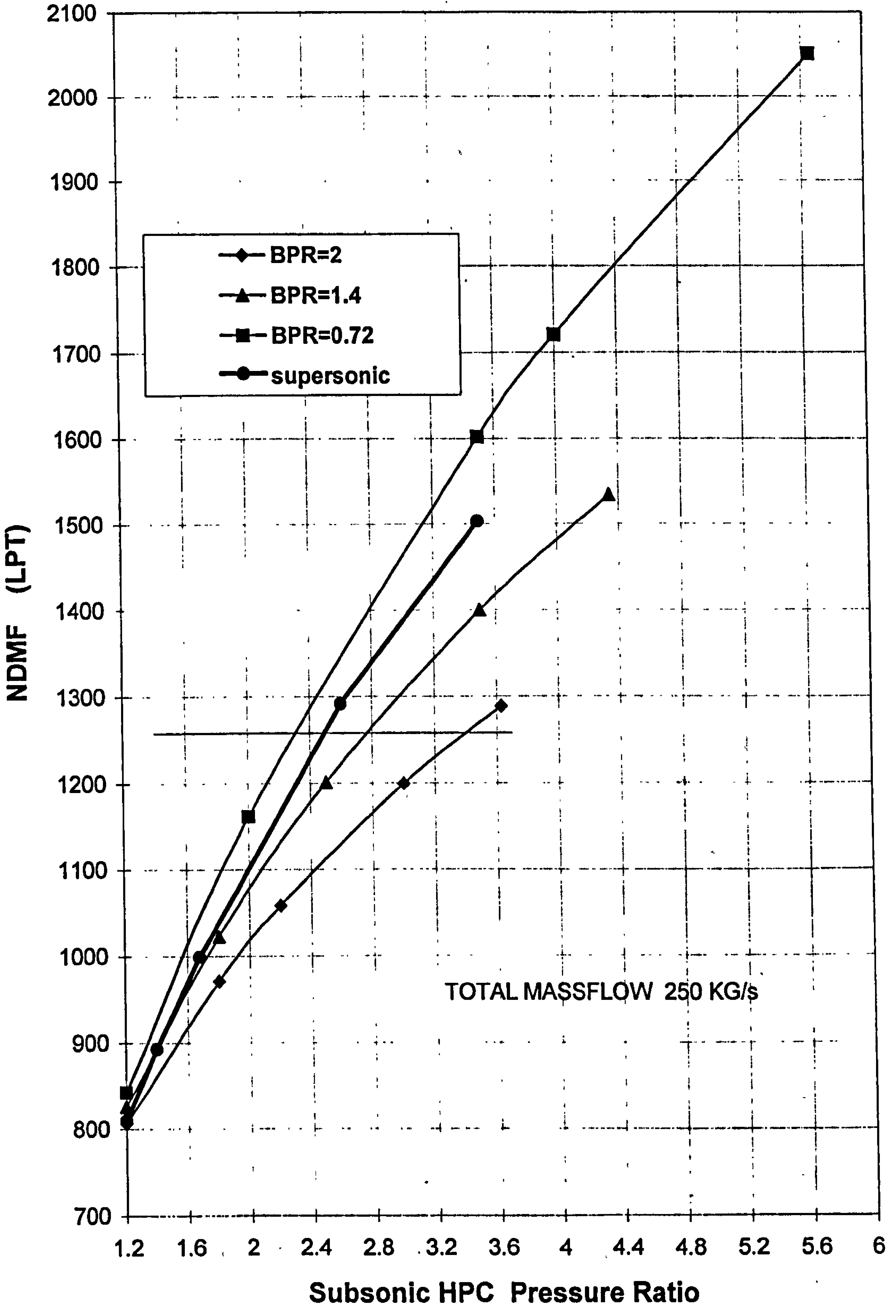


FIGURE 6.6 Effect Of HPC Pressure Ratio On LPT Non-Dimensional Mass Flow For Different Total Mass Flows

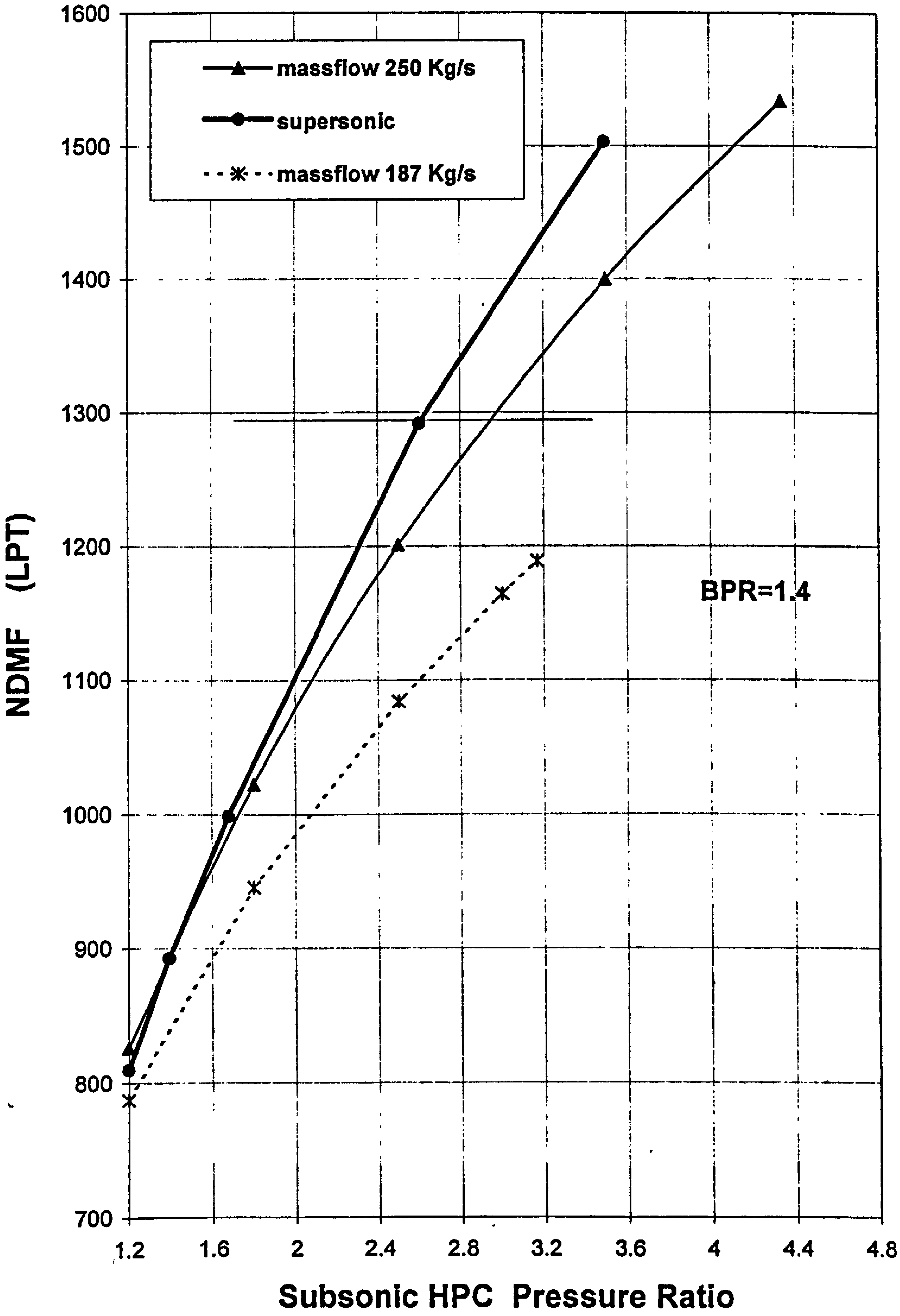


Figure 6.8 Variations Of BPR, OPR, SFC And The Specific Thrust In Function of The LPT NGVs Angles.

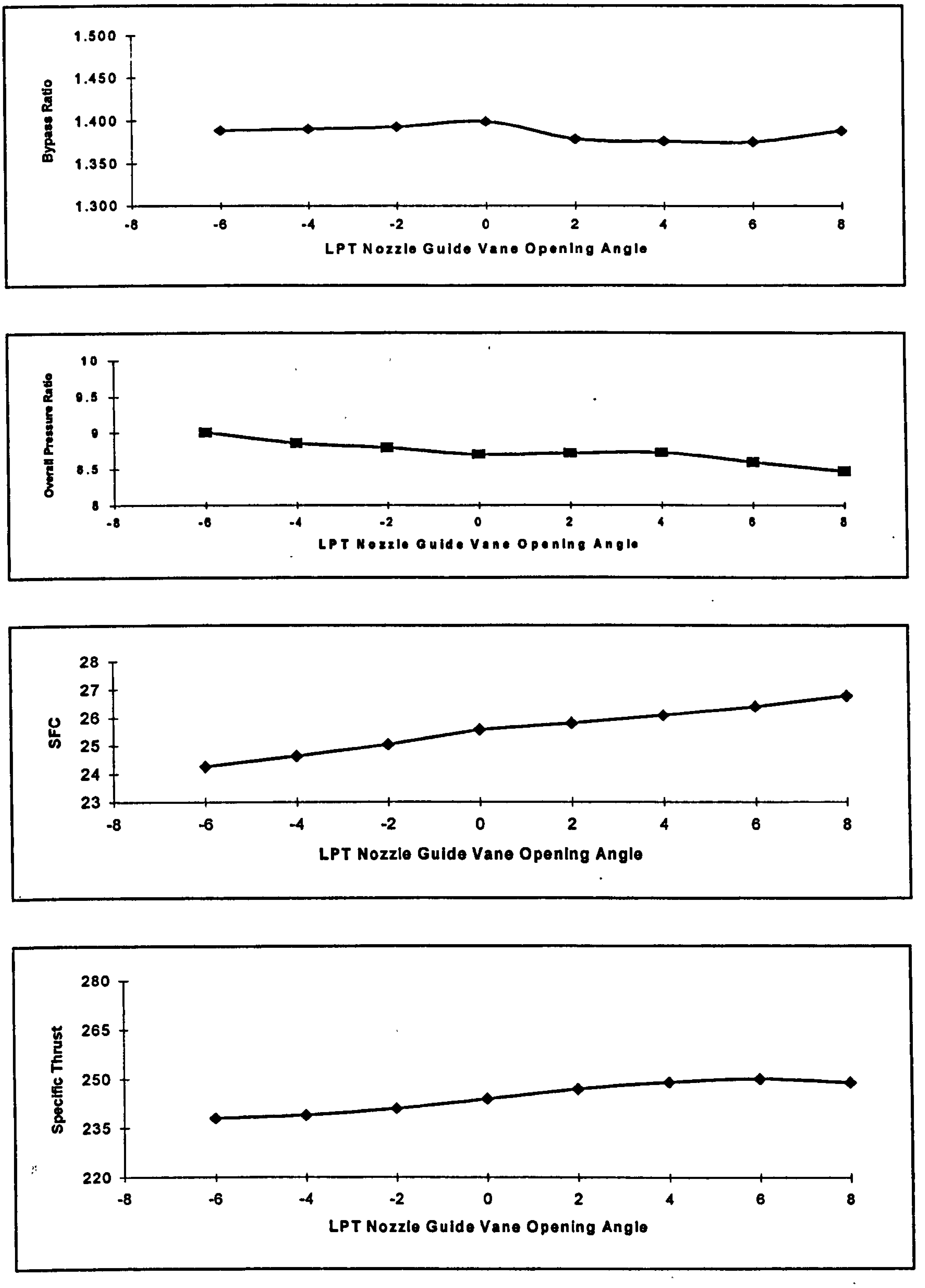


Figure 6.9 Effect of Opening And Closing The LPT NGVs Angles On Component Pressure Ratios

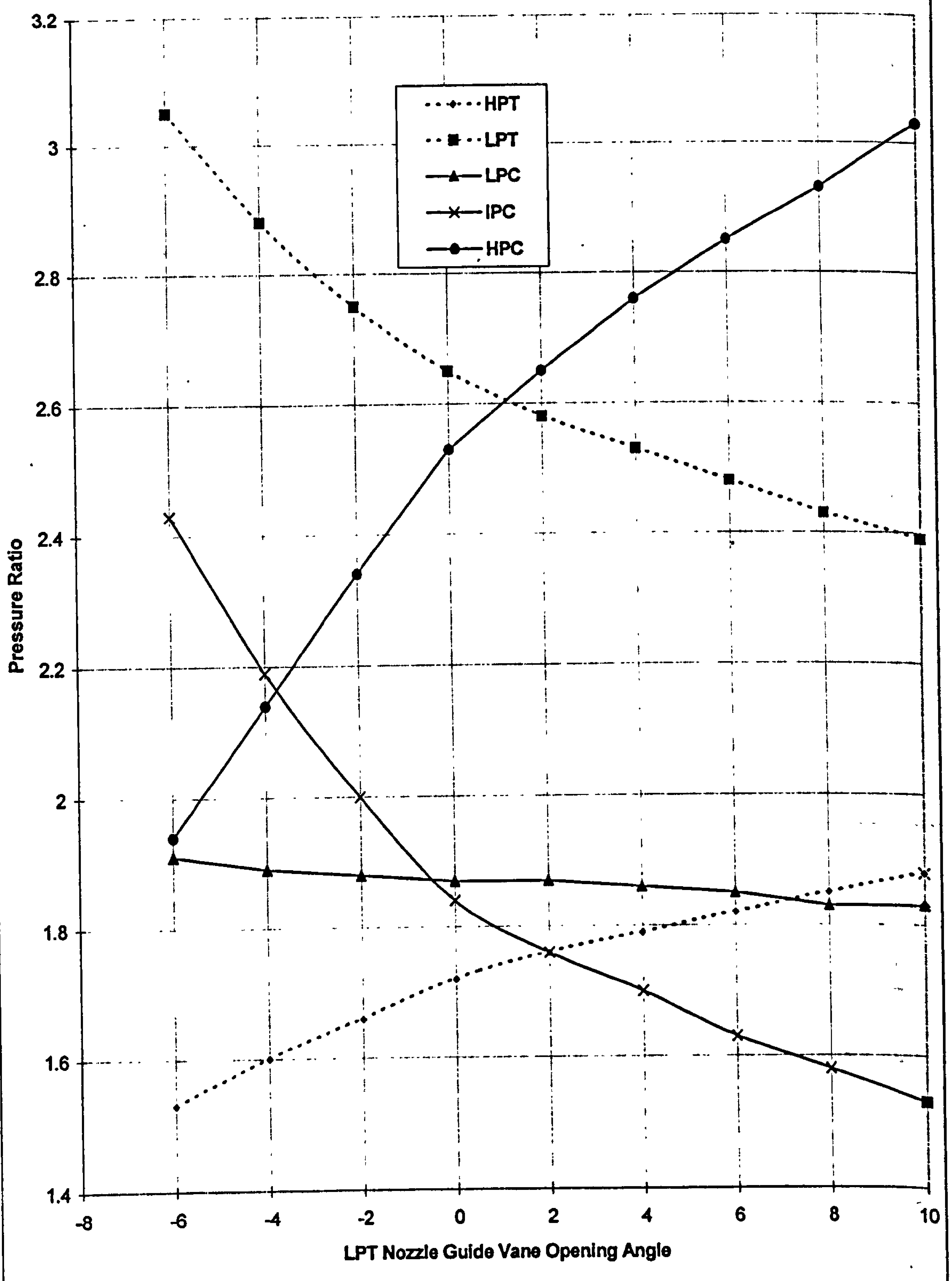
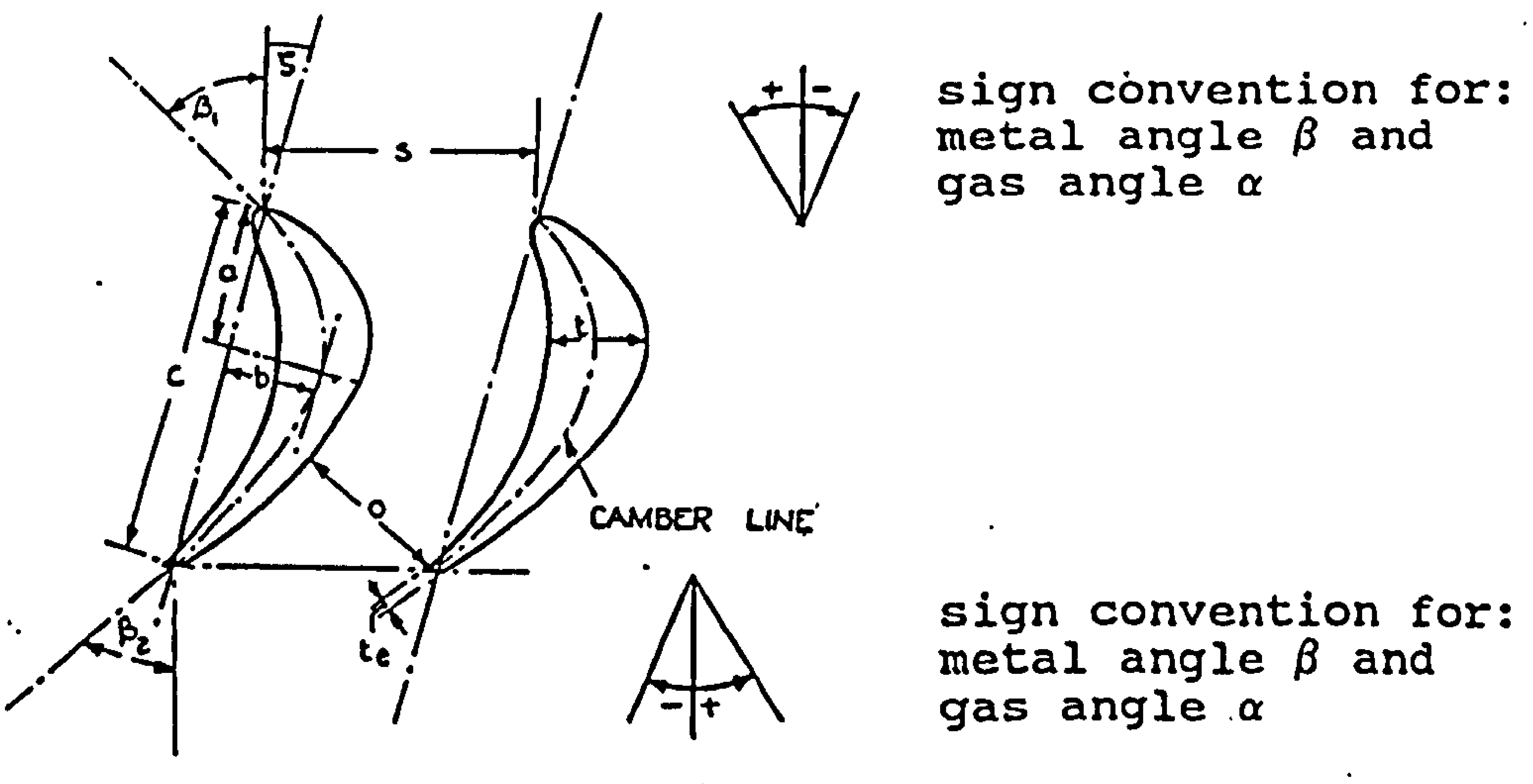


FIGURE 6.11 Turbine Blade Nomenclature



a = Distance of point of max camber from L.E.
b = Maximum camber
c = Chord
e = Mean radius of curvature of upper blade surface between throat and T.E. ($e=j^2/8Z$)
o = Blade opening
s = Blade pitch
t = Maximum thickness
t_e = Trailing edge thickness
β = Blade angle
ξ = Stagger angle

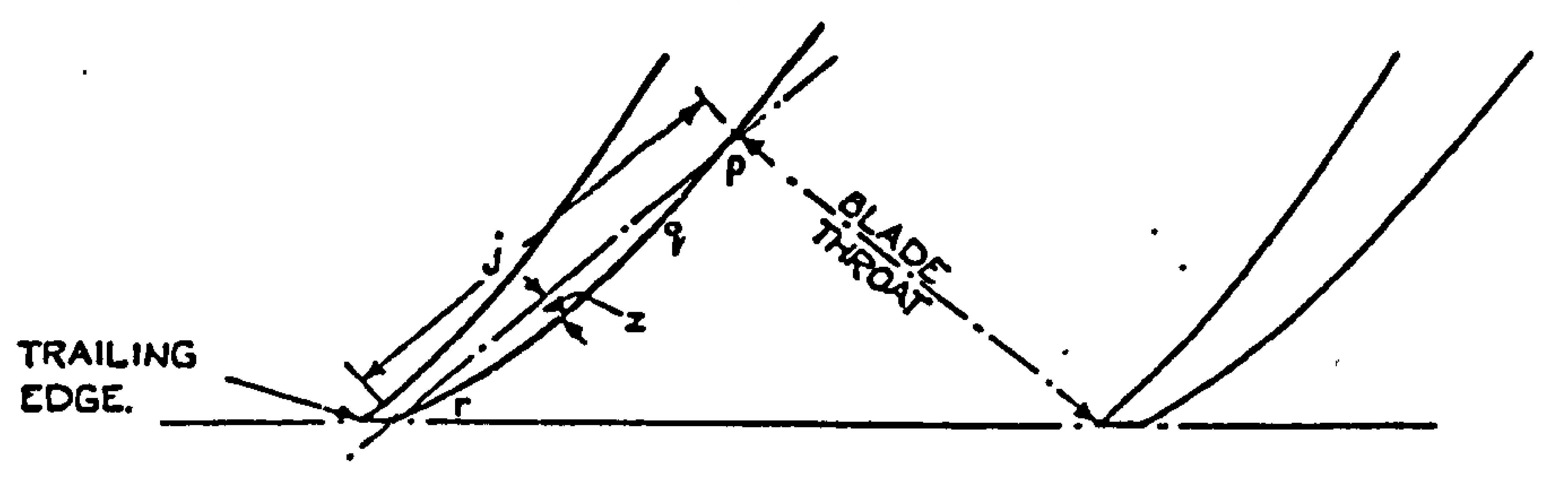


FIGURE 6.12 Turbine Blade Angles

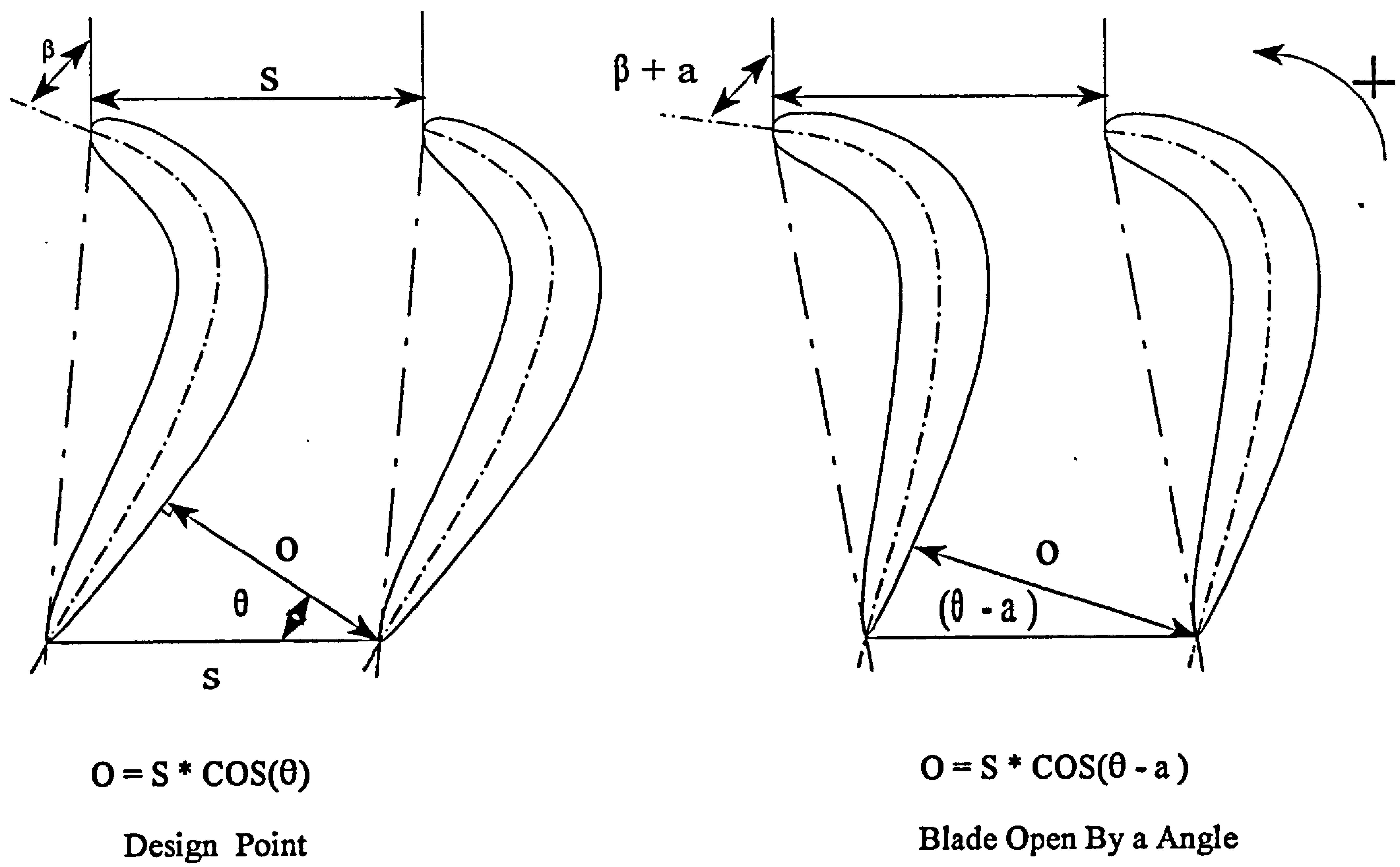
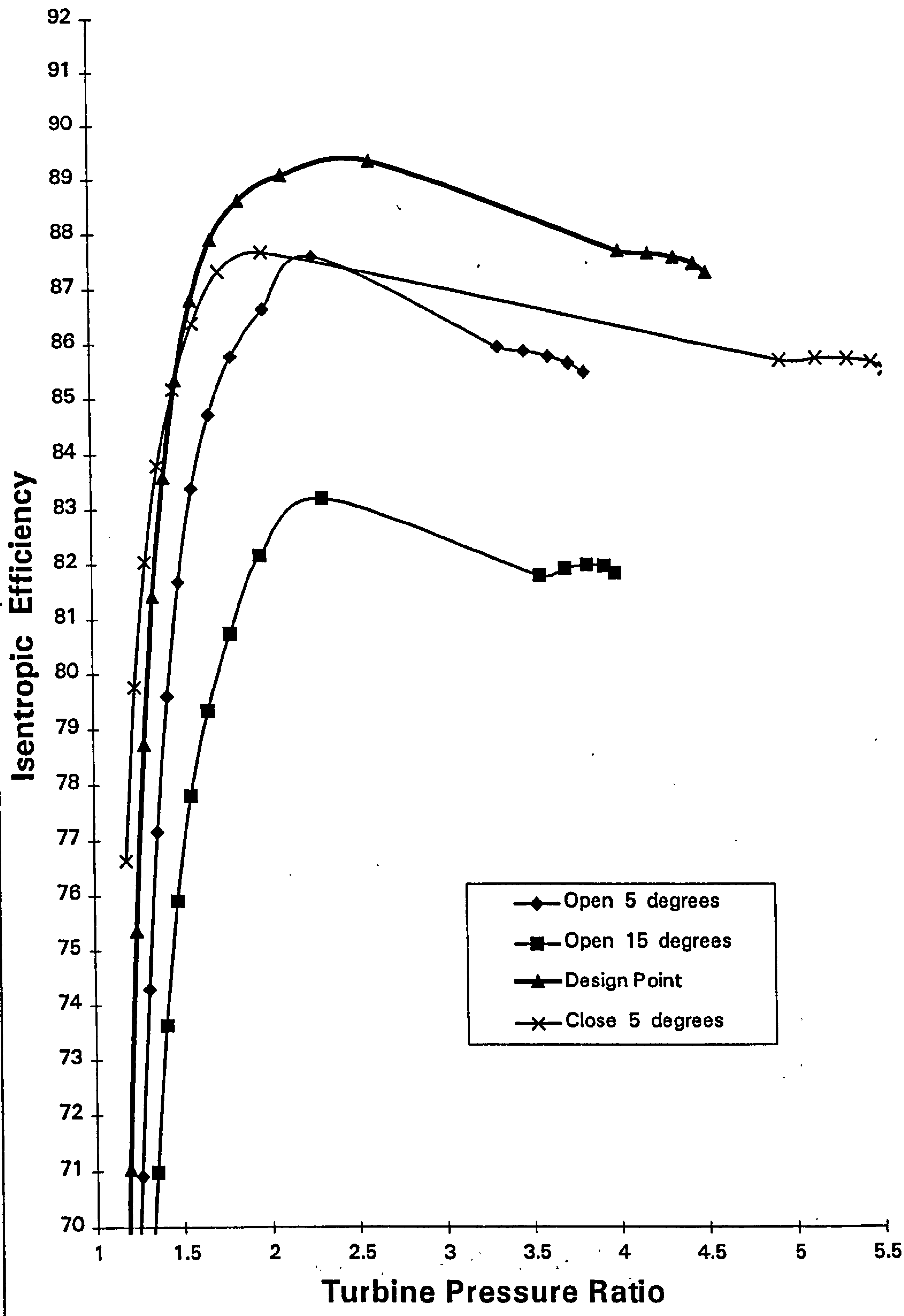


FIGURE 6.13 Effect of Variable Geometry on L.P. Turbine Efficiency



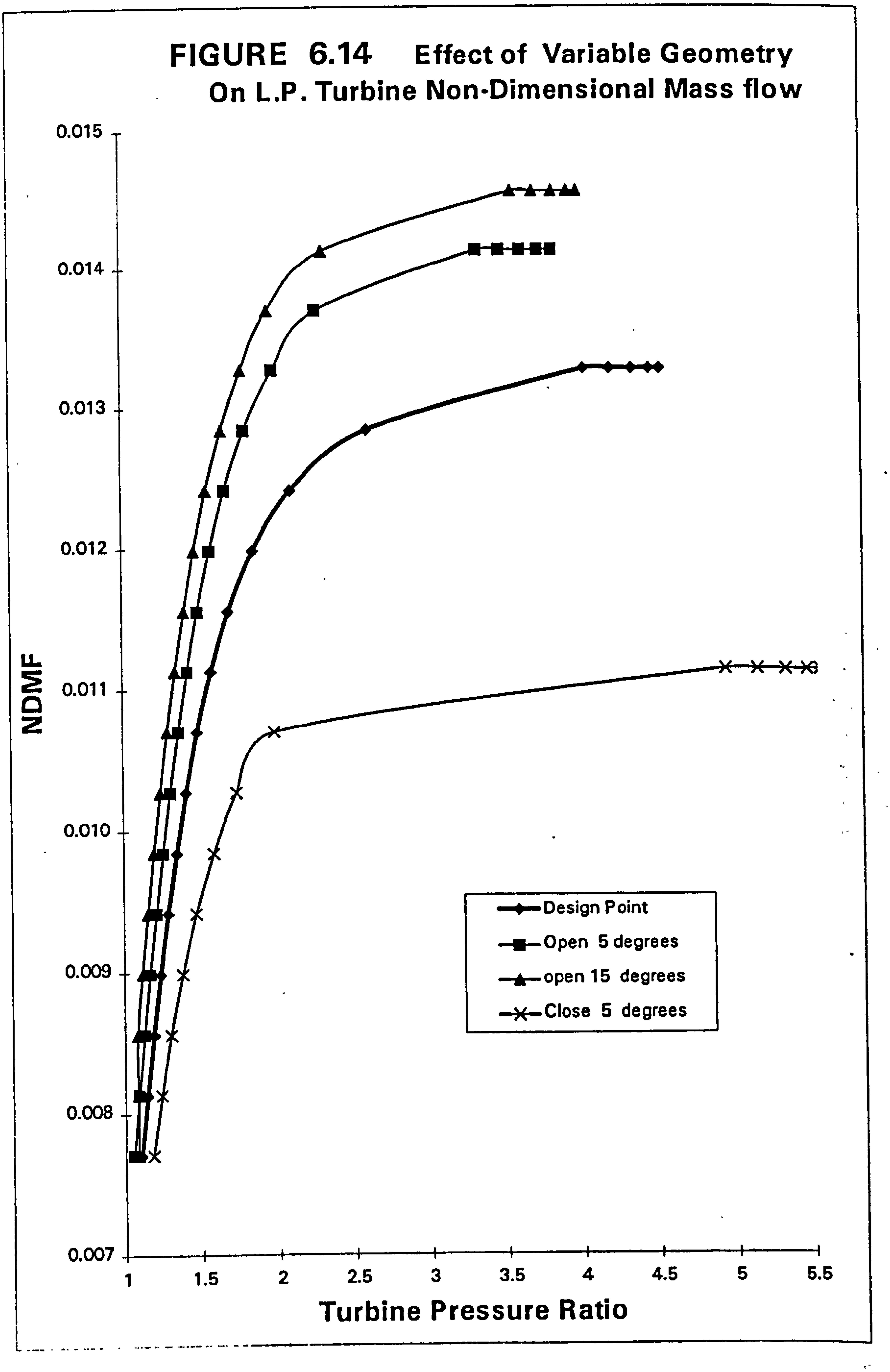


FIGURE 6.14 Effect of Variable Geometry
On L.P. Turbine Non-Dimensional Mass flow
NDMF
0.015
0.014
0.013
0.012
0.011
0.010
0.009
0.008
0.007
1
1.5
2
2.5
3
3.5
4
4.5
5
5.5
Turbine Pressure Ratio
Design Point
Open 5 degrees
open 15 degrees
Close 5 degrees

CHAPTER 7

Conclusion

High speed civil transports could be feasible and may represent a significant new product opportunity for manufacturers and airlines as long as promising technologies pan out. The fundamental question is still to be answered!...How the VCE will meet the noise regulation at take-off while maintaining an economic performance at subsonic and supersonic cruises?

CHAPTER 7

Conclusion

7.1 Conclusions

An updated version of the Turbomatch program was modified and tested. Now, this new version can be used to study variable cycle engines, especially to simulate the transition from one mode to another. It can use variable geometry compressors, turbines and variable convergent divergent nozzle. Real compressor and turbine maps can be used by the program in order to obtain a more accurate result. Examples of these applications were given. By using this new version, the bypass ratio of a turbofan engine can be varied up to a 100% relative to the design value depending on the extreme flight mach numbers. This is one of the novelties of the new version of the Turbomatch program.

One of the main topics investigated in this work was the overall uninstalled performance of three variable cycle engine concepts for future SSTs. The fuel bill, for two standard missions, was estimated as well. The results of this study indicated that the TFTJ and MTF engines are quite similar in terms of general suitability. The Mid-Tandem fan engine appears to be an attractive proposition from the point of view of sizing. However, this comes with a small penalty in fuel consumption.

The DBE was found inadequate to a Mach of 2.7 where the ram pressure ratio is two high. Originally, the fact that the bypass stream is split after the IP compressor is to increase the bypass pressure ratio associated with the HP mode. For the LP mode this bypass stream is split after the LP compressor in order to reduce the bypass pressure ratio. This engine layout was found unnecessary for a Mach 2.7 . This was reflected in difficulties matching the two cycles and this lead to supersonic and subsonic bypass ratios lower than the other two engines. The alternative to the DBE is the TFTJ engine where the bypass ratio is split after the LP compressor in both modes.

The study of the effect of installation was undertaken in order to push the comparison between the engines a step forward. The airframe engine integration analysis consisted of estimating the nacelle friction drag, the pre-entry drag due to the intake, the afterbody drag due to the nozzle and the wave drag due to the shock waves at the supersonic cruise. A sizing calculation was carried out for the whole nacelle including the intake and the nozzle. The maximum length and diameter for each engine was obtained. The MTF engine was the smallest in terms of diameter and length (1.75 m x 16.7 m). The DBE was the biggest (1.97 m x 18.9 m).

The installation investigation indicated that the MTF has the lowest nacelle drag due to its lowest diameter. The air friction drag was the most dominant part in the total drag at supersonic cruise because the intake was designed in such a way that the pre-entry, afterbody and wave drags were minimum. At the subsonic cruise, the percentage of pre-entry drag was higher due to the mismatch between the intake and the engine characteristics.

The installed performance showed that the TFTJ and the MTF engines are still quite similar in terms of fuel consumption and general suitability. However, the MTF appears to have a better position than before. This result is not surprising because most of the installation effect was carried out at the design points where the intake and nozzle design was optimised. The effect of installation at climb and acceleration was not really assessed, this needs the full knowledge of the nacelle and engine designs and the schedule of operations at off-design which is outside the scope of this work.

The uninstalled performance studies showed that the best supersonic bypass ratio, in terms of size and fuel consumption, is around 0.7 . The installation effects showed that the best overall performance is obtained for a bypass ratio of 0.5 . A few iterations supported by wind tunnel test are needed in order to select the best bypass ratio at supersonic mode.

In the search of improving the engine fuel consumption at the subsonic mode, the use of variable geometry in the low-pressure turbine was investigated at this mode. It was revealed that SFC, OPR and the BPR at the subsonic point are not significantly affected when opening or closing the nozzle guide vanes. These parameters are only a function of the high-pressure turbine matching which has a fixed geometry.. On the other hand, the use of variable geometry in the low-pressure turbine would make the matching between the cycles easier by offering more choices in selecting the high-pressure compressor pressure ratio at the subsonic mode. Hence, more flexibility is obtained to optimise the fan pressure ratio and improve the surge margin control.

For the first time at Cranfield University, the effect of opening and closing the nozzle guide vanes on the turbine off-design characteristics was obtained. This was done by manually changing some parameters in the input file for the Turboff program.

7.1.1 Over Sizing

The LP compressor inlet cross section area is an important factor in determining the size of the engine. If the engine is to be optimised at the supersonic design point the inlet LP compressor cross section area is 1.5527 sq. meter , Where T_{in} = 530 K, P_{in} = 1.3 atm for a standard intake, the inlet mass flow necessary to achieve the required thrust is 268 kg/sec, assuming the compressor face maximum inlet Mach number is 0.5 and a Hub/Tip ratio of 0.4.

At take-off conditions where (P_{in} = 1 atm, T_{in}= 288 K) the maximum mass flow able to pass through the above area is 280 kg/sec where 453 kg/sec is needed to meet the noise regulations at this point. The area required to pass the above mass flow is 2.4984, it is 60% higher than the area required for the supersonic cruise. This over sizing will considerably increase the weight of the engine and may offset some of the advantages gained by using a variable cycle engine. Therefore, other studies need to be focused on how to meet the noise regulations at take-off without considerably over sizing the engine. This problem is, somehow, resolved by the concept of the MTF engine where the extra mass flow needed at take-off is passed through lateral intakes to the Fan which is located in such a way that the over sizing of the engine would be minimum. A more detailed study of this concept is needed in order to assess its full potential. .

7.1.2 Intake

A mixed compression inlet was designed and optimised for the supersonic cruise, the intake pressure recovery estimated at this point is around 0.93, where a value of 0.84 was used by the Turbomatch program which uses a standard intake. It has been found that the intake had to have variable geometry at the subsonic mode in order to reduce the pre-entry drag due to the spillage at this point, where the engine airflow required is much less than the intake can deliver. The length and therefore the weight of the intake is a function of the supersonic flight Mach number, the intake length increases when the flight Mach number increases. This should be taken into consideration when choosing the future supersonic SST aircraft.

7.1.3 Variable Geometry Compressor

The extensive use of variable geometry was investigated in the LP and IP compressors. Varying the compressor stator angles will play an important role in changing the cycle bypass ratio in the two modes. The running lines on their variable geometry maps were obtained for the different compressors. In general, the surge margins are acceptable on the LP and IP compressor maps despite the awkward shape of some of the running lines. The efficiency levels of the Fan for the MTF and the LP compressor for the DBE are not acceptable (around 65%) and further investigations are needed. On the other hand, the efficiency levels for the IP compressors are satisfactory.

The component efficiencies in design and off-design will play an important role in the selection of the future SST powerplant. Especially the compressor variable stators where they could vary up to 30° relative to design point for the LP compressor and up to 10° for the IP compressor. The effect of changing the stators angles on the efficiency should be analysed in more detail for the LP and IP compressors.

7.1.4 Variable Convergent Divergent Nozzle

A convergent divergent nozzle which uses variable throat and exit areas was used successfully, the throat and exit areas were independent of each other. The throat area has an important role in controlling the LP compressor surge margin while the cycle changes are taking place. The exit area is changed in order to obtain a full expanded nozzle in almost all the flight envelope, this will maximise the net thrust obtained from the engine. However, at the supersonic point, the exit area reaches its maximum value which is approximately equal to the maximum nacelle cross section area. This area is less than the one which leads to a full expansion exit and the nozzle is under expanded, a small penalty in SFC had to be paid.

An examination of the nozzle throat area changes requirements showed that the variable geometry requirement for all the engines is approximately similar, with the largest requirement being that of the MTF. As far as the nozzle exit area is concerned, the DBE has the largest requirement, of the order of 300%. This is due to the large change in mass flow going throw the core engine when the cycle change from one mode to another.

7.1.5 Other Remarks

An examination of the mixer area variation reveals that apparently there is a large requirement for variable geometry. However, because the mixer is placed at a large engine diameter the change in flow area could be easily achieved

Many recent studies revealed that the future supersonic civil transport will fly at a Mach between 2 and 2.4 maximum. Reducing the flight supersonic Mach number from 2.7 to around 2 will considerably facilitate the matching between the cycles and reduce the constraints imposed on the LP compressor and other components and the chance of feasibility will increase.

7.2 Contributions

The main contributions of this work can be summarised in the three following points:

A- The new version of the Turbomatch program was modified and tested in order to simulate and study variable cycle engines. This included, for the first time at Cranfield University, the simultaneous use of variable stator compressor, variable mixing area and variable convergent divergent nozzle in one input file. Three different variable cycle jet engine concepts for future SSTs were simulated and analysed. Variable geometry compressors were designed and their maps were obtained. The matching method used in previous work was refined and adapted for supersonic civil transport.

B- The foundations of the calculation of the installed performance were laid down in chapter 5. The whole design of the nacelle and major components was carried out and this gave a real idea about the dimensions of the future engine which will power the future SST aircraft. Some problems and difficulties in matching the different components were discovered, underlined and discussed. This work is the genesis for further studies about variable cycle engines for future supersonic civil transport.

C- Finally, equally important, was the investigation of using variable geometry in the low-pressure turbine and its consequences on the matching between the cycles. Furthermore, the effects of opening and closing the nozzle guide vanes on the turbine off-design characteristics were obtained for the first time at Cranfield University.

7.3 Further Work

The present work shows the uninstalled and installed performance of three concepts of variable cycle engines for future SSTs. It showed that the three engines are quite similar in terms of general suitability. Further works on this subject should contain the following points:

1- Pushing the comparison between the engine a further step forward by comparing the weights and mechanical design and integrity of each engine.

2- Analysis of the effect of LP turbine variable geometry for the DBE which had some problems in matching the cycles for the two modes. This effect was not significant for the TFTJ engine, it depends on the engine layout.

3- Modifying the turbine software available in the department in order to incorporate the effect of opening and closing the guide vanes on the turbine off-design characteristics.

4- As far as the variable geometry compressors are concerned, the Turbomatch program should be changed in order to be able to select in its COMPRE bricks a non zero value for the stator angle at the design point. For example, at the supersonic point the LPC stator angles are closed by 30°, Turbomatch program should be able to select this point as the design point. This is not the case at the moment where a zero angle should always be selected at the design point. This will make the Turbomatch program a very powerful tool to simulate VCE.

5- The mechanism of opening and closing the mixing area should be studied and analysed in a detailed manner in order to assess the full potential of this technique.

6- Assessment of the use of variable geometry in general and the use of variable stator angle in particular on the compressor and turbine efficiencies. Moreover, the effect of compressor and turbine efficiencies changes on SFC should be analysed.

7- Further studies need to be focused on the key issue of how to meet the noise regulation at take-off without over sizing the engine at other points of the flight envelope.

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APPENDIX A

TURBOMATCH PROGRAM MODIFICATIONS

CONVERGENT-DIVERGENT NOZZLE

1 Introduction

The convergent-divergent nozzle brick "NOZDIV" is one of the few components that has some degree of variable geometry for off-design performance calculations.

At the present stage the convergent-divergent nozzle has three modes of variable geometry operation available for the user. These modes of operation can be selected by a switch in the NOZDIV brick(Data). The three modes are:

1) BD(1) = 1 : Nozzle throat and exit area are floating.
2) BD(1) = -1 : Nozzle throat and exit area are both fixed.
3) BD(1) = -2 : Nozzle throat is fixed and nozzle exit is floating.

This convergent-divergent nozzle uses a coefficient velocity map to calculate the real nozzle exit velocity after the losses introduced by angularity and friction. The velocity coefficient is a function of nozzle pressure ratio(total pressure/ambient pressure) and area ratio (exit area / throat area) Figure A.2. The maximum pressure ratio used in the map is 23 and the maximum area ratio is 2.15. For a given area ratio equal or inferior to 2.15 and pressure ratio equal or inferior to 23 the program can find the velocity coefficient relative to these parameters. If the pressure ratio is higher than 23 the program can extrapolate the curves to the new value. But if the area ratio is higher than 2.15 this extrapolation can not be done, and the velocity coefficient in this case corresponds to an area ratio equal to 2.15.

Unfortunately, the NOZDIV brick was not working properly, and a few problems had been discovered when testing the program (Reference 4 Chapter 6). Some of them are due to a wrong input file parameters when using the afterburner brick, especially the matching between errors and variables in the input file. Some of them are due to a few programming mistakes in "NOZDIV" subroutine.

In this chapter, we will see a general presentation of the convergent-divergent nozzle functions and a report of the problems found and their corrections. Also some improvements have been done in order to make this program more accurate and precise for the general use.

2 Convergent-Divergent Nozzle Capabilities

As it mentioned earlier, the convergent-divergent nozzle has three modes of operations. The first tow modes were introduced in the initial program written by Macmillan in 1974 (Ref. 2). The third mode was introduced by Van Den Hout in 1991 (Ref. 3). These modes are :

2.1 Fixed Nozzle Operation

The terms "fixed" and " floating" in Turbomatch have a special meaning. The term "fixed" means that the program will not change the value of the area during off-design runs of the particular engine unless they are changed manually by the user himself. This can be done by changing the value of this fixed areas, the throat area can be changed as brick data item (BD(2) of the NOZDIV brick) and the nozzle exit area can be changed as a station vector item.

This option is however the most difficult option because any change in afterburner or gas generator conditions can not be translated directly by the user into the necessary throat area change, also the change in exit area for full expansion is not readily known to the user. A trial and error procedure then becomes necessary. This option is not very useful when the user only wants optimum performance calculated during off-design runs.

2.2 Floating Nozzle Operation

The term "floating" means that the Turbomatch program will change the value of both the throat and exit area automatically during any off-design runs. The scheme will change the value of the throat area such that it can handle any change in non-dimensional mass flow due to changes in gas generator conditions or the operation of an afterburner. The nozzle exit area will float to a value such that full expansion of the gases will occur. This option is the easiest option since the user does not have to operate the nozzle manually and the program will automatically set the nozzle for maximum engine performance. But however the control of the surge margin of the L.P.compressor is poor in this option where the throat area has an important role on the surge margin.

2.3 Fixed Nozzle Throat And Floating Nozzle Exit

A more difficult situation will occur when the user wants to manipulate the throat area and at the same time wants the exit area to float to full expansion for maximum thrust. The user has to change both areas simultaneously, the throat area to the wanted value but the correct exit area for full expansion is not readily known to the user. This would then mean a trial and error procedure in which the exit area has to be changed several times in order to get the right conditions. To the user this new option would mean that the throat area can be changed manually and the exit area will automatically float to a position at which full expansion of the gases will occur.

When calculating the number of variable and errors according to Turbomatch Manual the user must remember that for each fixed nozzle of any type (convergent or convergent-divergent) an error is required. This error is thus also required for this option.

3 Problems And Corrections

According to Reference 4 (Chapter 6) two major problems were found :

A) The first one was reported as following '..... While we were working with an engine........, a fatal error for a program of this characteristics appeared. A value for a static pressure greater than the total pressure in the exit of the nozzle was obtained. This kind of errors should not be admissible. ...., and the only place we have found this error is in the condiv nozzles......'.

After checking the input file the origin of this error was found, it occurred because the number of errors was greater than the number of variables. The nozzle was switched to the position BD(1) = 1(nozzle throat and exit are floating) the numbers of errors and variables were balanced, after that the user decided to use afterburner and to switch the nozzle to the position BD(1) = -1 (throat and exit are fixed) the number of errors, in this case, has increased by one, and the user forgot to eliminate the error introduced in the compressor brick data in order to balance again the number of errors and variables. That is why the static pressure was greater than the total pressure.

When the calculations were made again but this time the compressor brick data number 5 was switched to 0 instead of 1 (so the numbers of variables and errors are equal) this fatal error did not occur. The origin of this error is related to the user and not to the program it self.

B) The second problem was that the nozzle exit area with the option (BD(1) = -2 throat area fixed and exit area floats) does not give the same results as if this same area was introduced manually in the option(BD(1) = -1 throat and exit areas are fixed). The origin of this problem is due to a programming mistake in the "NOZDIV" subroutine. The formula comparing the areas in this subroutine was:

```
If (ABS(AB-ABAD) . LE. 0)  ABAD=AB
```

It was replaced by :

```
If (ABS(AB-ABAD) . LE. 0.005)  ABAD=AB
```

This modification gave a compatible results for the two above options. Moreover, many corrections related to output messages have been done in order to obtain a more accurate results expected from this kind of programs.

4 Other Improvements

Two areas are concerned by these improvements, they are the velocity coefficient map and the formulas used to calculate the different parameters at the exit of the convergent-divergent nozzle.

4.1 Velocity Coefficient Map

The old and the new maps are shown in Figure A.1. The difference between them is due to the fact that the numbers found in turbomatch were not exactly the same found in the original map (Ref. 1). The numbers found in turbomatch were rounded to three numbers after the comma, for example the velocity coefficient 0.9784 in the original map will be 0.978 in turbomatch and that is why the old map was not smooth enough. The original map has been reintroduced in turbomatch to replace the old one.

Unfortunately, this map is old and not updated. However, it was very difficult to find a new map with the same range of area ratios and pressure ratios to replace the actual one. In the future this map need to be extended to a wider range of these parameters to meet the increasing need for investigating supersonic aircrafts.

The velocity coefficient need to be very precise, because this will affect directly the gross thrust. For example a 2% increase in velocity coefficient would result in a 2 % increase in design point gross thrust and the net thrust would be increased by 3.5 % approximately and the S.F.C. would decrease by 3.5 % . A change of 3.5 % in S.F.C. is significant. We see again the importance of the high precision values of the velocity coefficient.

4.2 Pressure Formulas

Turbomatch uses two different formulas to calculate the pressure. They are:

a- $p = \exp[S_t(T) - R.Ln(P) + S_s(t)]$. $S_t(T)$, $S_s(t)$ are the entropies relative to the total and static temperatures, P is the total pressure, R is the gas constant.

b- $p = P.(t/T)^{\gamma/(\gamma-1)}$.

When the nozzle pressure ratio is low these two formulas give the same results. However when the nozzle pressure ratio increases the difference between the two pressures increases and could reach 10 % . That is why the second formula has been replaced by the first one which is general and correct in all cases. This change had led to more accurate and consistent results. These replacements took place only in a few places in the subroutine NOZDIV where it was appropriate.

4.3 Some Curve Test

The following results have been carried out for a single spool turbojet at S.L.S conditions with TET = 1800 k. The variation of mach number and nozzle's areas are always related to the same engine and these same conditions .

A- Influence Of Mach Number On Net Thrust

Figure A.3 shows the influence of mach number on the net thrust for different configurations of the Convergent-Divergent nozzle. These configurations are:

1- Nozzle fully expanded.
2- The ratio (nozzle exit area/throat area) is fixed and equal 2.15 .
3- The (exit static pressure/ambient pressure) is fixed and equal 2.
4- Using a Convergent nozzle.

Configurations 2 and 3 show that the convergent-divergent nozzle is overexpanded and underexpanded respectively. The effect of these overexpansion and underexpansion on the net thrust is shown in Figure A.3.

B- Influence Of Exit Area On The Net Thrust

Figure A.4 shows the effect of changing the exit area on the net thrust, the throat area is fixed. We can see the old curve where the different formulas, discussed above, were used to calculate the static pressure. For the fully expanded nozzle the program was using the first formula, for other cases the second formula was used.

However we can see that the maximum thrust is achieved just before the full expansion of the nozzle (i.g. static pressure equal to the ambient pressure). This result contradicts the known theory of that the maximum thrust is achieved when the nozzle is fully expanded. This difference is due to the velocity coefficient map. To prove this result the same curve was plotted, but with the velocity coefficient equal to one(without losses), here the maximum thrust is achieved very close to the full expansion position. The difference is in the range of precision expected from Turbomatch program.

C- Influence of Throat Area on Net Thrust

Figure A.5 shows the influence of the throat area on the net thrust, S.F.C. and specific thrust. We can see that there is an optimum throat area where the net thrust is maximum and the S.F.C. is minimum.

Figure A.1 Velocity Coefficient Maps

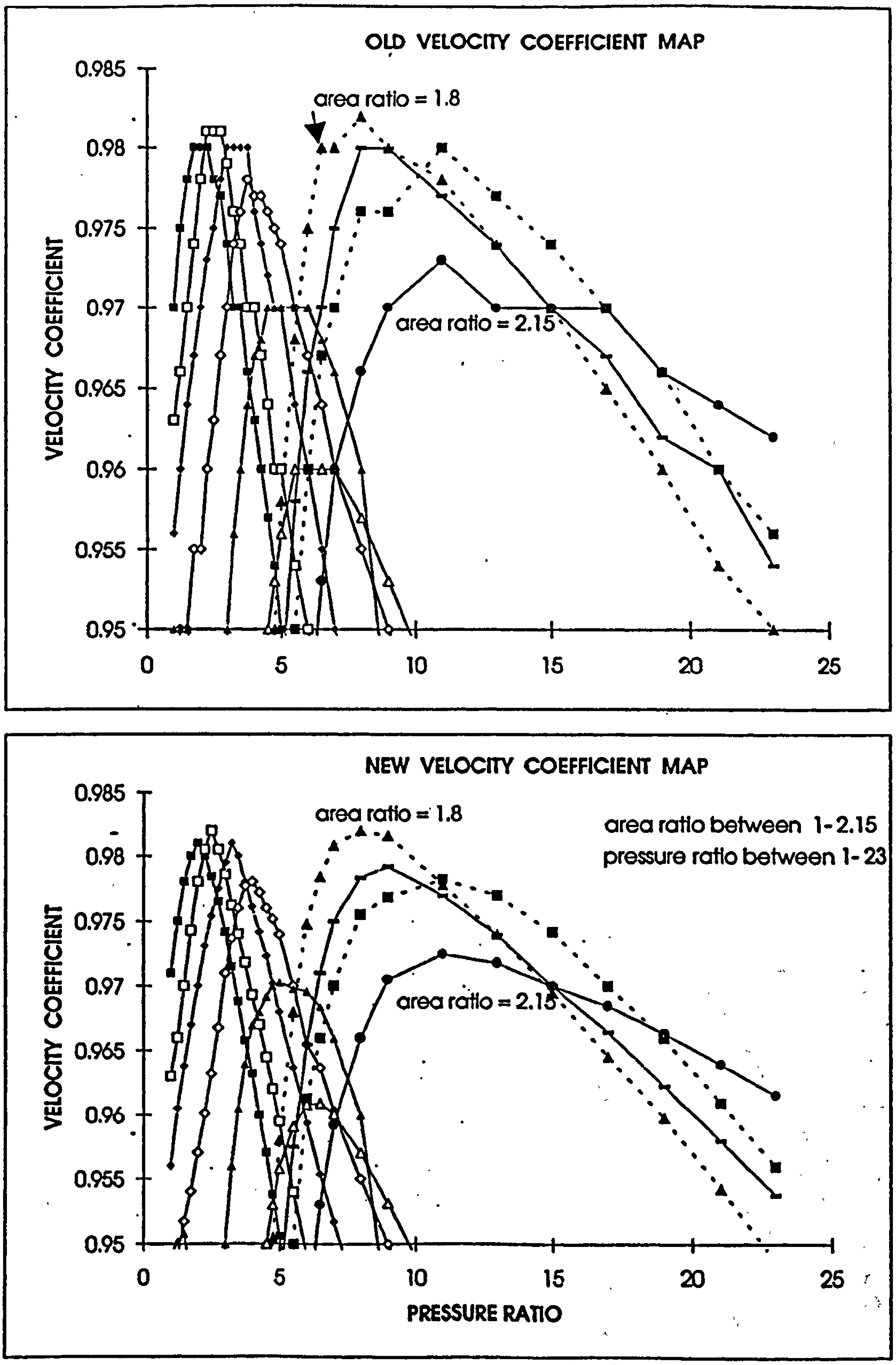


Figure A.2 New Velocity Coefficient Map

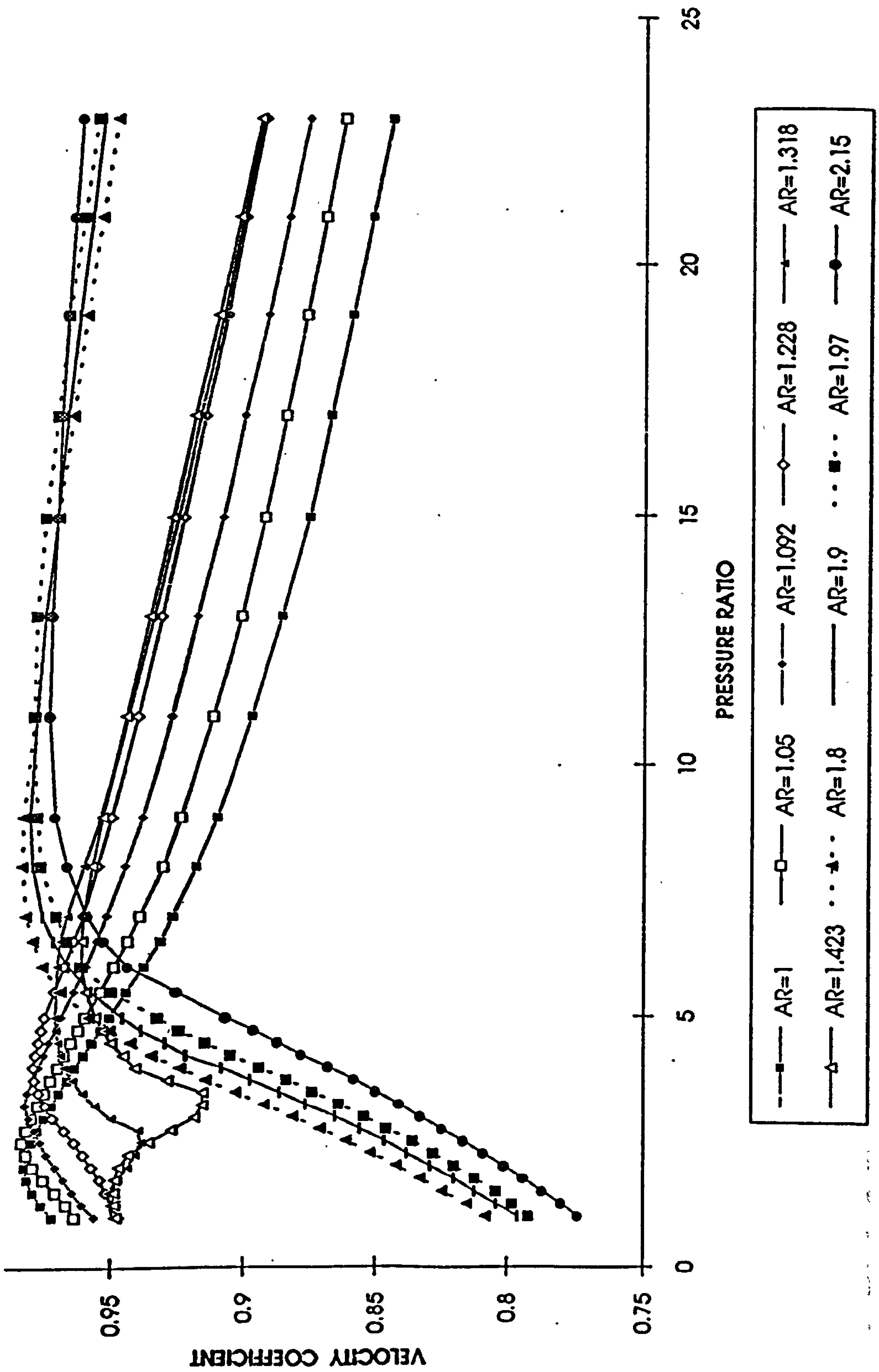


Figure A.3 Influence Of Mach Number On Net Thrust

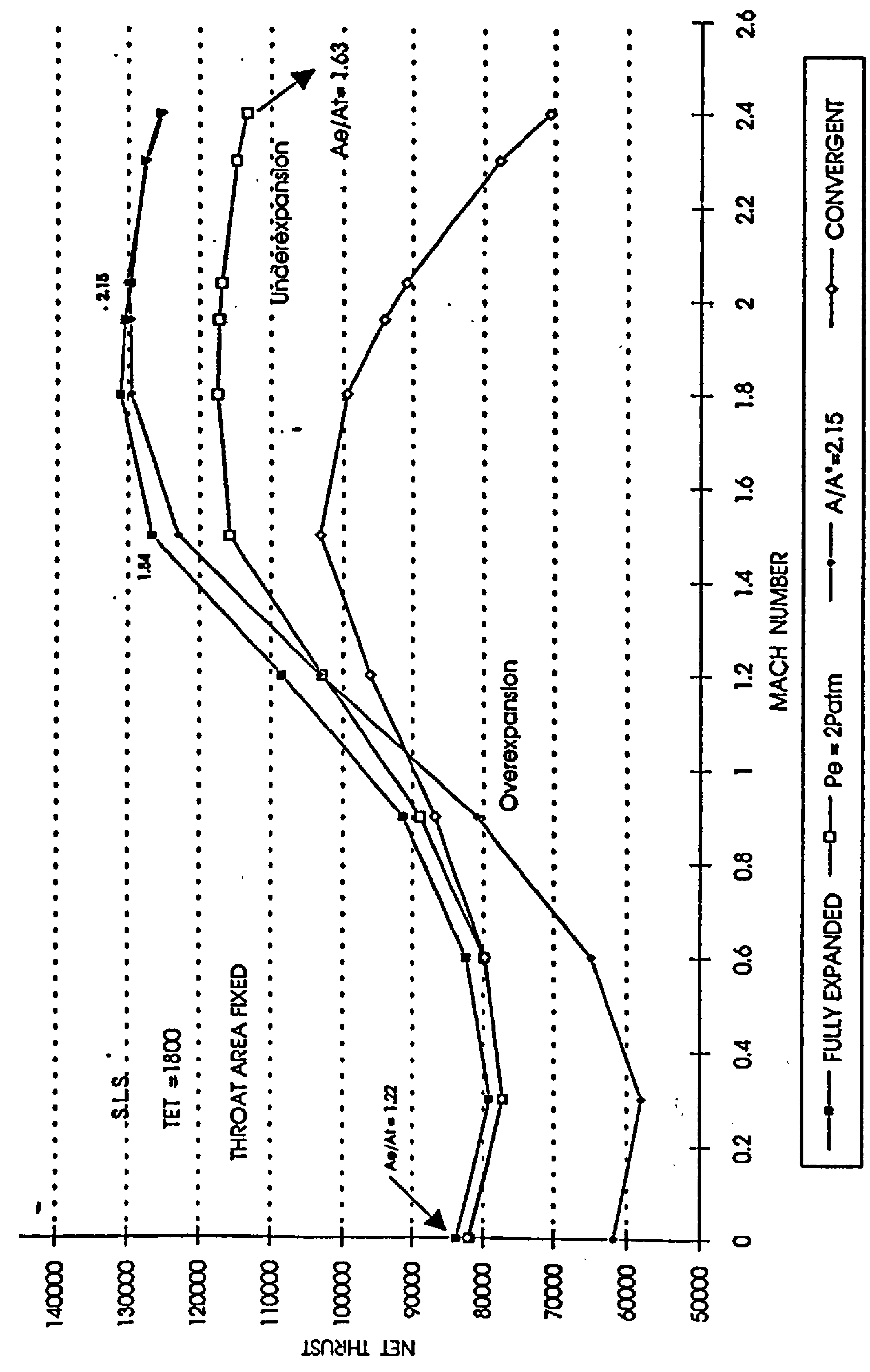


Figure A.4 Influence Of Exit Area On Net Thrust

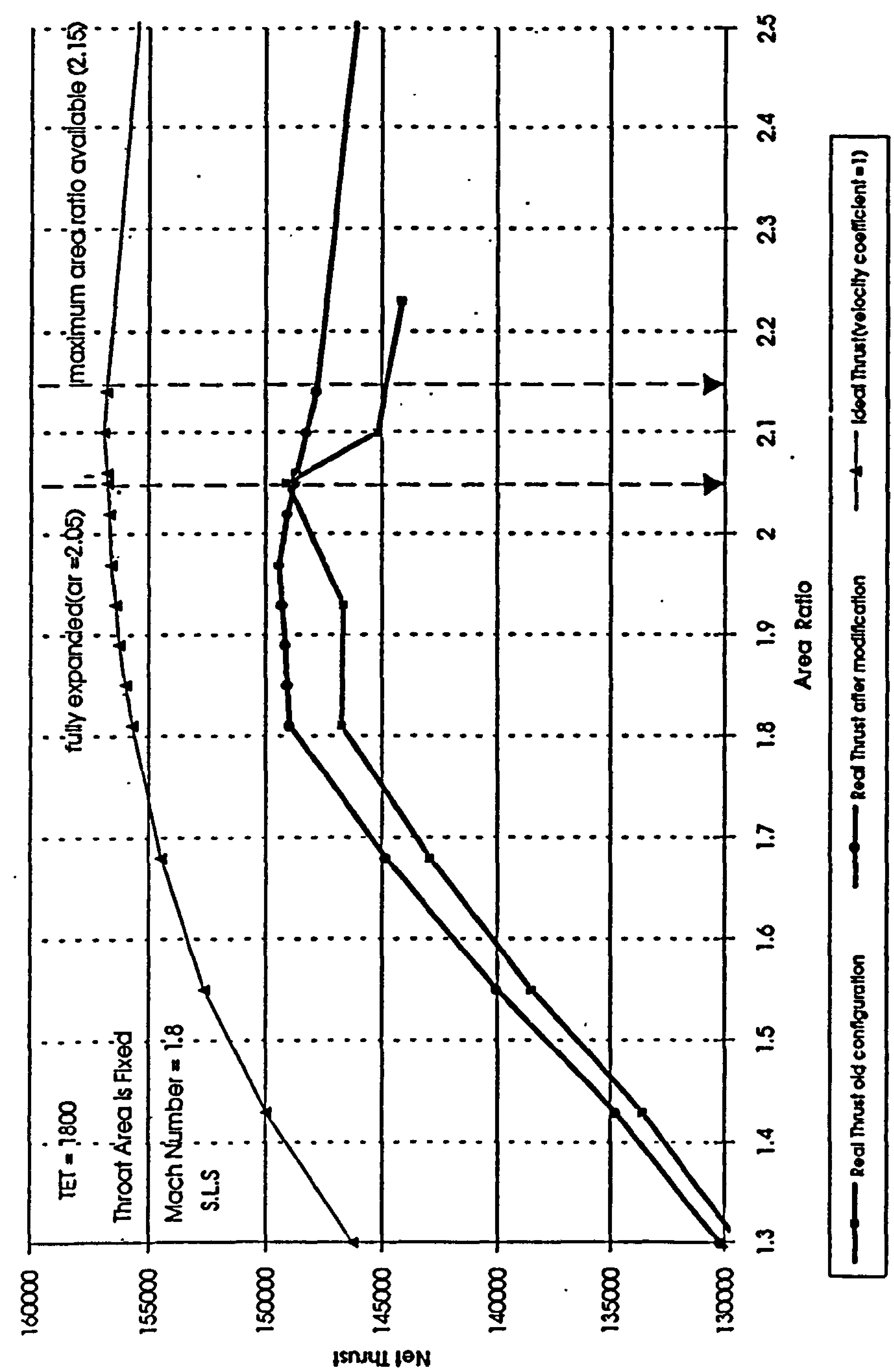
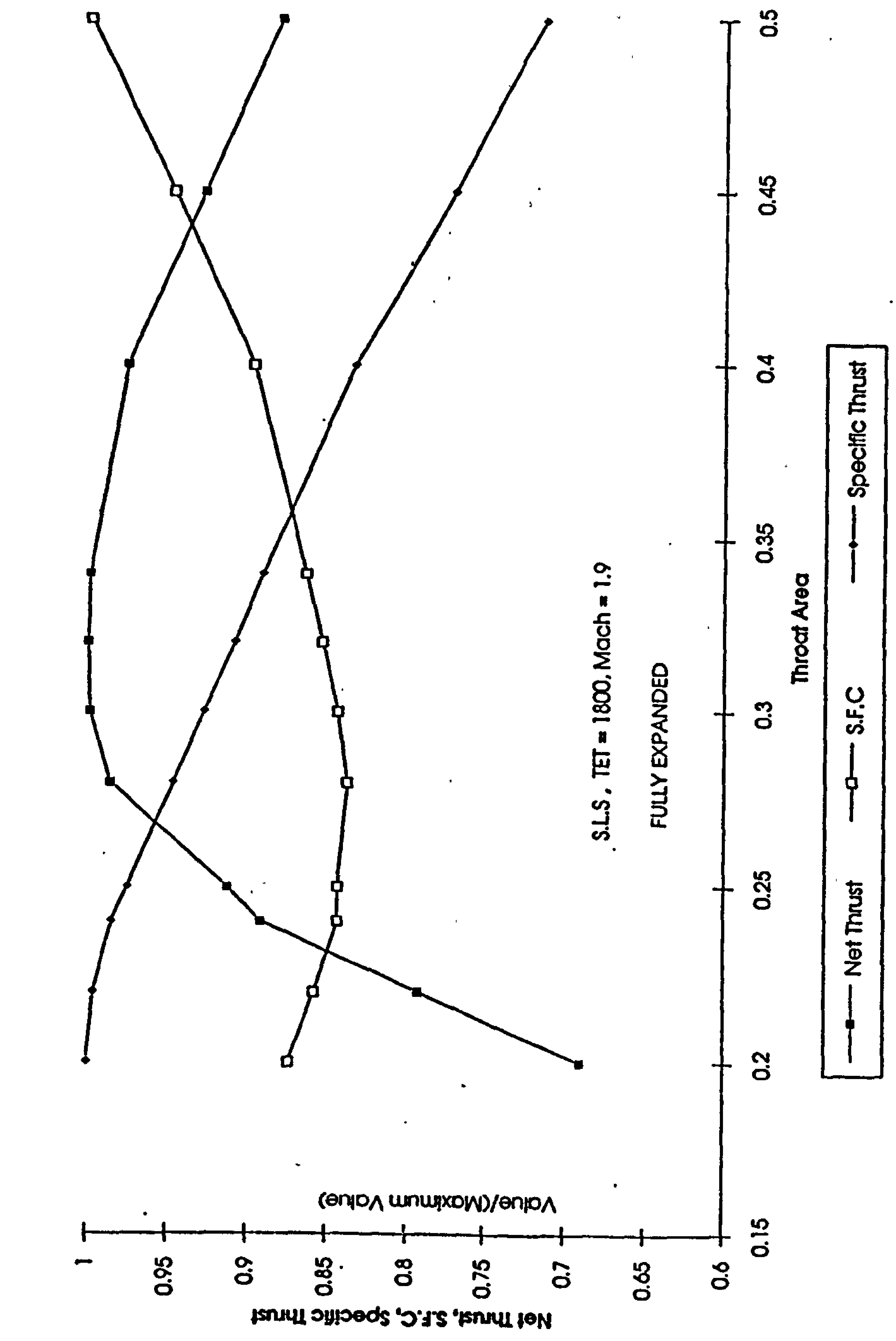


Figure A.5 Influence Of Throat Area On Net Thrust



PREMAS BRICK

PREMAS brick is used to split the flow in two streams. It is used usually after the L.P. compressor to split the flow into the bypass stream and the core stream or it was used to split the cooling mass flow from the main stream. It was necessary to use this brick always after one of the compressors, it was not possible to use this brick directly after the intake. For some applications, it is necessary to split the mass flow just after the intake. For this raison it was decided to modify the program allowing it to use PREMAS directly after the INTAKE brick.

The following lines were added to the "MASTER.FOR" file where PREMAS brick is called:

```
860    IF(ICW(I-1,1).EQ.1.AND.DES.EQ.-1) THEN
               IF (SC.EQ.0) THEN
                       SV(SA,2)=SV(SB,2)
               ELSE
                       SV(SA,2)=SV(SB,2)+SV(SC,2)
               ENDIF
                       D(2)=SV(SB,2)-SV(SA,2)*D(1)
                       SV(SA-1,2)=SV(SA,2)
       ENDIF
```

The use of this option is not frequent. It was tested on one simple example. However, it need to be test on a wider range of applications.

DUCTER BRICK

A small error was found in the DUCTER subroutine at the "Bricksb.for" file: The ducter pressure loss at off-design calculation was always zero whatever the input brick data value. The correction will allow the pressure losses to be accounted at off-design calculations. The two following lines:

```
DLP=DELP*PA
IF(IDES.LE.0) DLP=CA(NDUCT)*WA*WA*TA/PA
```

were replaced by:

```
CA(NDUCT)=DELP*PA*PA/(WA*WA*TA)
DLP=DELP*PA
IF(IDES.LE.0) DLP=CA(NDUCT)*WA*WA*TA/PA
```

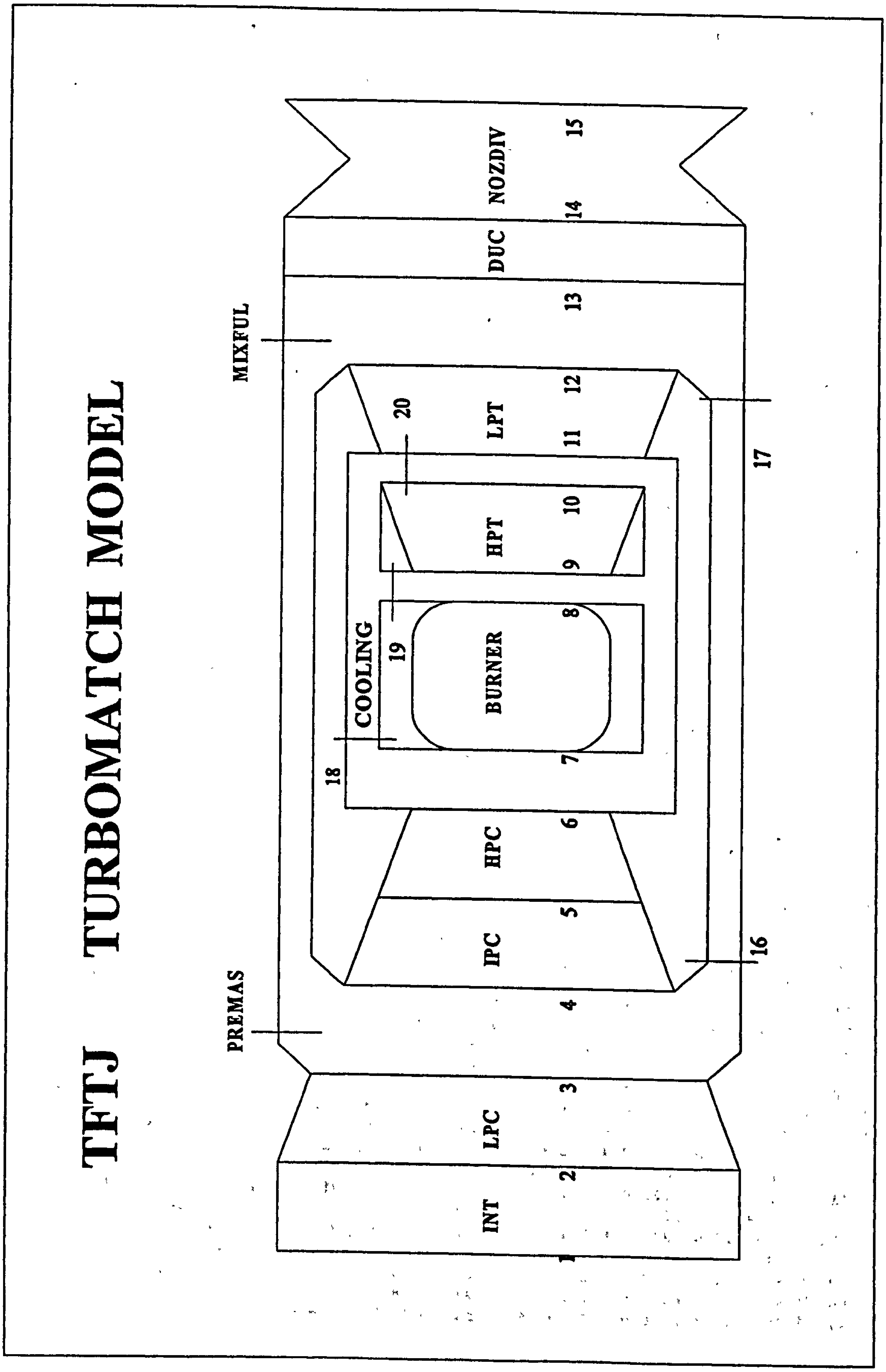
Program To Insert Variable Compressor Maps Into The Turbomatch Input File

```
      program varcom
      character*80 eff,char,stringe,stringc,TURINP,TUROUT,MNAME
      REAL DATAE1,DATAE2,DATAE3,DATAC1,DATAC2,DATAC3,DATA,DATE
      REAL R(15),S(3),RPM,CN(10),RPMDP,MPME(10,15)
      INTEGER NUMBER,ANGLE,L,AN(6)
      CHARACTER YN
C FILES NAME ***************************************************
      WRITE(6,*)'IS IT A VARIABLE GEOMETRY COMPRESSOR (Y/N)'
      READ(6,'(A1)') YN
      open(3,FILE='PRCMEF.DAT',STATUS='UNKNOWN')
      IF((YN.EQ.'Y').OR.(YN.EQ.'y')) L=6
      IF((YN.EQ.'N').OR.(YN.EQ.'n')) L=1
      WRITE(6,*)'GIVE THE RATIO (RPM DP/ MAXIMUM RPM ON THE MAP)'
      READ(6,*) RPM
      DO 777 ANGLE=1,L
      WRITE(6,*)'FILE NUMBER ',ANGLE,' OF ',L
      write(6,*)'give efficiency input file name'
      read(6,'(a80)') eff
      write(6,*)'give characteristics input file name'
      read(6,'(a80)') char

C OPENNING FILES ***********************************************
      open(1,FILE=eff,STATUS='UNKNOWN')
      READ(1,'(A)',END=999) STRINGE
      open(2,FILE=char,STATUS='UNKNOWN')
      READ(2,'(A)',END=999) STRINGC

      J=0
      N=10
      DO 5 I=1,15
      R(I)=0
5      CONTINUE
10      continue
      read(1,'(A)',end=999) STRINGE
      read(2,'(A)',end=999) STRINGC
      READ(STRINGE,'(E15.6,2F15.6)')DATAE1,DATAE2,DATAE3
      READ(STRINGC,'(E15.6,2F15.6)')DATAC1,DATAC2,DATAC3
      DATA=DATAC1/(288.15)**0.5*101325
      DATAE=DATAE2/100
      S(3)=DATAC2
      S(2)=DATA
      S(1)=DATAE
      DO 20 I=1,3
      R(16-(J+I))=S(I)
20      CONTINUE
      J=J+3
      IF(J.GT.12) THEN
      J=0
      DO 22 I=1,15
      MPME(N,I)=R(I)
22      CONTINUE
      IF(N.EQ.10) RPMDP=DATAC3*RPM
      CN(N)= DATAC3/RPMDP
```

```
      WRITE(6,*) N,RPMDP,DATAC3,CN(N)
      N=N-1
      ENDIF
      GOTO 10

999   CONTINUE
      close(1)
      CLOSE(2)
      DO 24 N=1,10
      IF(MPME(N,2).LT.100) THEN
      WRITE(3,'(1X,F4.2,F4.0,F6.3,4(F5.2,F4.0,F6.3))',ERR=999)(MPME(N
     1,I),I=1,15)
      ELSE
       WRITE(3,'(1X,F4.2,F5.0,F6.3,4(F5.2,F5.0,F6.3))',ERR=999)(MPME(N
     1,I),I=1,15)
      ENDIF
24    CONTINUE
777   CONTINUE
      CLOSE(3)

      write(6,*)'give TURBOMATCH input file name'
      read(6,'(a80)') TURINP
      open(1,FILE=TURINP,STATUS='UNKNOWN')
      write(6,*)'give TURBOMATCH OUTPUT file name'
      read(6,'(a80)') TUROUT
      open(2,FILE=TUROUT,STATUS='UNKNOWN')
25    READ(1,'(A)',END=889) STRINGC
      WRITE(2,'(A)',ERR=889) STRINGC

      IF ((STRINGC(1:1).EQ.'O'.AND.STRINGC(2:2).EQ.'P'.AND.STRINGC(3:3)
     1.EQ.' ').OR.(STRINGC(1:1).EQ.'D'.AND.STRINGC(2:2).EQ.'P'.AND.STRI
     1NGC(3:3).EQ.' '))THEN
      write(6,*)'give NUMBER OF MAP TO INSERT'
      read(6,*) NUMBER
      WRITE(2,'(1X,I1)',ERR=889) NUMBER
      IF((YN.EQ.'Y').OR.(YN.EQ.'y')) THEN
      WRITE(6,*)'GIVE SIX VALUES OF THE RALATIVE TO DP STATOR ANGLE'
      READ(6,'(A80)') EFF
      WRITE(2,'(A)',ERR=888) EFF
      ENDIF
      WRITE(2,'(1X,F4.2,9(F5.2))',ERR=888) (CN(I),I=1,10)
      open(3,FILE='PRCMEF.DAT',STATUS='UNKNOWN')
30    READ(3,'(A)',END=888) STRINGE
      WRITE(2,'(A)',ERR=889) STRINGE
      GOTO 30

888   CONTINUE
      CLOSE(3)
      ENDIF

      GOTO 25
889   continue
      CLOSE(1)
      CLOSE(2)
      end
```

APPENDIX B

TURBOFAN-TURBOJET ENGINE


TFTJ TURBOMATCH MODEL
PREMAS
MIXFUL
INT
LPC
IPC
HPC
COOLING
BURNER
HPT
LPT
DUC
NOZDIV
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15
16
17
18
19
20

TURBOMATCH INPUT FILE

Turbofan-Turbojet Engine

```
TFTJ.TAKEOFF////
OP SI KE VA FP
3
-10 -5 0 5 5 5
 0.50 0.60 0.70 0.75 0.80 0.85 0.90 0.95 1.00 1.10
 1.00 77. 0.028 1.03 74. 0.307 1.10 68. 0.742 1.18 58. 0.897 1.20 47. 0.814
 1.12 84. 0.688 1.16 82. 0.770 1.22 76. 0.871 1.28 68. 0.899 1.30 58. 0.824
 1.28 93. 0.852 1.32 92. 0.878 1.37 87. 0.899 1.41 79. 0.889 1.42 70. 0.833
 1.36 98. 0.874 1.41 97. 0.890 1.45 92. 0.896 1.48 85. 0.881 1.49 78. 0.834
 1.44 103. 0.877 1.50 102. 0.889 1.53  98. 0.888 1.56  92. 0.871 1.57  85. 0.831
 1.52 109. 0.872 1.59 107. 0.881 1.62 104. 0.877 1.64  99. 0.859 1.65  93. 0.825
 1.60 114. 0.861 1.68 113. 0.871 1.72 110. 0.866 1.74 106. 0.848 1.75 100. 0.817
 1.69 120. 0.850 1.78 119. 0.859 1.82 117. 0.854 1.84 113. 0.837 1.85 108. 0.810
 1.77 125. 0.833 1.88 124. 0.844 1.93 123. 0.840 1.96 120. 0.829 1.96 118. 0.811
 1.87 131. 0.770 2.01 131. 0.789 2.09 131. 0.793 2.13 131. 0.791 2.16 130. 0.787
 0.97 75. 0.497 1.01 73. 0.094 1.09 66. 0.717 1.17 56. 0.908 1.20 45. 0.820
 1.09 82. 0.618 1.13 80. 0.733 1.20 74. 0.872 1.27 65. 0.911 1.29 55. 0.829
 1.23 91. 0.838 1.28 89. 0.876 1.34 84. 0.910 1.39 76. 0.904 1.41 67. 0.838
 1.31 95. 0.875 1.37 94. 0.898 1.42 89. 0.912 1.46 82. 0.898 1.48 73. 0.840
 1.40 100. 0.890 1.46  99. 0.904 1.50  94. 0.909 1.54  88. 0.890 1.56  80. 0.839
 1.48 106. 0.891 1.55 104. 0.902 1.59 100. 0.901 1.63  94. 0.880 1.64  87. 0.836
 1.55 111. 0.884 1.64 110. 0.895 1.68 106. 0.892 1.72 101. 0.872 1.74  94. 0.833
 1.63 117. 0.876 1.73 116. 0.887 1.78 112. 0.882 1.82 107. 0.864 1.84 102. 0.830
 1.71 121. 0.862 1.84 120. 0.876 1.90 118. 0.871 1.93 114. 0.855 1.95 110. 0.829
 1.81 128. 0.818 2.03 128. 0.840 2.12 127. 0.837 2.17 126. 0.829 2.19 126. 0.821
 0.95 73. 1.594 0.98 71. 0.287 1.07 64. 0.679 1.16 55. 0.916 1.19 43. 0.823
 1.06 80. 0.503 1.10 78. 0.679 1.18 72. 0.868 1.26 63. 0.919 1.28 53. 0.829
 1.19 88. 0.811 1.25 86. 0.863 1.31 81. 0.915 1.38 73. 0.911 1.40 64. 0.836
 1.27 93. 0.861 1.33 91. 0.893 1.39 86. 0.919 1.44 78. 0.906 1.47 70. 0.840
 1.34 97. 0.884 1.41 96. 0.906 1.46 91. 0.918 1.52 85. 0.901 1.54 77. 0.848
 1.42 103. 0.893 1.50 101. 0.908 1.55  97. 0.913 1.60  91. 0.896 1.62  84. 0.853
 1.50 108. 0.892 1.58 107. 0.905 1.64 103. 0.906 1.68  98. 0.891 1.71  91. 0.855
 1.57 113. 0.885 1.67 112. 0.899 1.73 109. 0.899 1.78 105. 0.885 1.80  99. 0.856
 1.63 118. 0.874 1.78 117. 0.891 1.84 115. 0.890 1.88 111. 0.878 1.91 107. 0.857
 1.73 124. 0.838 1.96 124. 0.863 2.05 124. 0.862 2.09 123. 0.856 2.12 122. 0.850
 0.95 73. 1.594 0.98 71. 0.287 1.07 64. 0.679 1.16 55. 0.916 1.19 43. 0.823
 1.06 80. 0.503 1.10 78. 0.679 1.18 72. 0.868 1.26 63. 0.919 1.28 53. 0.829
 1.19 88. 0.811 1.25 86. 0.863 1.31 81. 0.915 1.38 73. 0.911 1.40 64. 0.836
 1.27 93. 0.861 1.33 91. 0.893 1.39 86. 0.919 1.44 78. 0.906 1.47 70. 0.840
 1.34 97. 0.884 1.41 96. 0.906 1.46 91. 0.918 1.52 85. 0.901 1.54 77. 0.848
 1.42 103. 0.893 1.50 101. 0.908 1.55  97. 0.913 1.60  91. 0.896 1.62  84. 0.853
 1.50 108. 0.892 1.58 107. 0.905 1.64 103. 0.906 1.68  98. 0.891 1.71  91. 0.855
 1.57 113. 0.885 1.67 112. 0.899 1.73 109. 0.899 1.78 105. 0.885 1.80  99. 0.856
 1.63 118. 0.874 1.78 117. 0.891 1.84 115. 0.890 1.88 111. 0.878 1.91 107. 0.857
 1.73 124. 0.838 1.96 124. 0.863 2.05 124. 0.862 2.09 123. 0.856 2.12 122. 0.850
 0.95 73. 1.594 0.98 71. 0.287 1.07 64. 0.679 1.16 55. 0.916 1.19 43. 0.823
 1.06 80. 0.503 1.10 78. 0.679 1.18 72. 0.868 1.26 63. 0.919 1.28 53. 0.829
 1.19 88. 0.811 1.25 86. 0.863 1.31 81. 0.915 1.38 73. 0.911 1.40 64. 0.836
 1.34 97. 0.884 1.41 96. 0.906 1.46 91. 0.918 1.52 85. 0.901 1.54 77. 0.848
 1.42 103. 0.893 1.50 101. 0.908 1.55  97. 0.913 1.60  91. 0.896 1.62  84. 0.853
 1.50 108. 0.892 1.58 107. 0.905 1.64 103. 0.906 1.68  98. 0.891 1.71  91. 0.855
 1.57 113. 0.885 1.67 112. 0.899 1.73 109. 0.899 1.78 105. 0.885 1.80  99. 0.856
 1.63 118. 0.874 1.78 117. 0.891 1.84 115. 0.890 1.88 111. 0.878 1.91 107. 0.857
 1.73 124. 0.838 1.96 124. 0.863 2.05 124. 0.862 2.09 123. 0.856 2.12 122. 0.850
 0.95 73. 1.594 0.98 71. 0.287 1.07 64. 0.679 1.16 55. 0.916 1.19 43. 0.823
 1.06 80. 0.503 1.10 78. 0.679 1.18 72. 0.868 1.26 63. 0.919 1.28 53. 0.829
 1.19 88. 0.811 1.25 86. 0.863 1.31 81. 0.915 1.38 73. 0.911 1.40 64. 0.836
 1.27 93. 0.861 1.33 91. 0.893 1.39 86. 0.919 1.44 78. 0.906 1.47 70. 0.840
```

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1.34 97. 0.884 1.41 96. 0.906 1.46 91. 0.918 1.52 85. 0.901 1.54 77. 0.848
1.42 103. 0.893 1.50 101. 0.908 1.55  97. 0.913 1.60  91. 0.896 1.62  84. 0.853
1.50 108. 0.892 1.58 107. 0.905 1.64 103. 0.906 1.68  98. 0.891 1.71  91. 0.855
1.57 113. 0.885 1.67 112. 0.899 1.73 109. 0.899 1.78 105. 0.885 1.80  99. 0.856
1.63 118. 0.874 1.78 117. 0.891 1.84 115. 0.890 1.88 111. 0.878 1.91 107. 0.857
1.73 124. 0.838 1.96 124. 0.863 2.05 124. 0.862 2.09 123. 0.856 2.12 122. 0.850
2
-5 5 0 5 5 5
0.50 0.60 0.70 0.75 0.80 0.85 0.90 0.95 1.00 1.10
1.05 37. 0.493 1.10 36. 0.670 1.19 33. 0.839 1.28 28. 0.832 1.31 22. 0.719
1.22 42. 0.801 1.27 41. 0.840 1.36 38. 0.868 1.45 33. 0.826 1.48 27. 0.729
1.41 48. 0.863 1.49 47. 0.879 1.58 44. 0.874 1.66 39. 0.825 1.69 34. 0.743
1.51 52. 0.873 1.62 51. 0.886 1.72 47. 0.875 1.80 43. 0.827 1.82 37. 0.751
1.63 56. 0.879 1.77 54. 0.890 1.87 51. 0.876 1.95 47. 0.830 1.97 41. 0.759
1.77 60. 0.884 1.93 59. 0.893 2.04 56. 0.878 2.12 51. 0.834 2.15 46. 0.768
1.90 64. 0.885 2.10 63. 0.895 2.23 60. 0.880 2.33 56. 0.839 2.36 51. 0.778
2.05 69. 0.885 2.30 68. 0.896 2.45 65. 0.882 2.56 61. 0.845 2.60 56. 0.788
2.17 73. 0.879 2.53 72. 0.894 2.71 69. 0.881 2.83 66. 0.847 2.87 62. 0.798
2.32 77. 0.844 3.07 77. 0.867 3.28 77. 0.856 3.39 76. 0.844 3.45 76. 0.832
1.05 37. 0.493 1.10 36. 0.670 1.19 33. 0.839 1.28 28. 0.832 1.31 22. 0.719
1.22 42. 0.801 1.27 41. 0.840 1.36 38. 0.868 1.45 33. 0.826 1.48 27. 0.729
1.41 48. 0.863 1.49 47. 0.879 1.58 44. 0.874 1.66 39. 0.825 1.69 34. 0.743
1.51 52. 0.873 1.62 51. 0.886 1.72 47. 0.875 1.80 43. 0.827 1.82 37. 0.751
1.63 56. 0.879 1.77 54. 0.890 1.87 51. 0.876 1.95 47. 0.830 1.97 41. 0.759
1.77 60. 0.884 1.93 59. 0.893 2.04 56. 0.878 2.12 51. 0.834 2.15 46. 0.768
1.90 64. 0.885 2.10 63. 0.895 2.23 60. 0.880 2.33 56. 0.839 2.36 51. 0.778
2.05 69. 0.885 2.30 68. 0.896 2.45 65. 0.882 2.56 61. 0.845 2.60 56. 0.788
2.17 73. 0.879 2.53 72. 0.894 2.71 69. 0.881 2.83 66. 0.847 2.87 62. 0.798
2.32 77. 0.844 3.07 77. 0.867 3.28 77. 0.856 3.39 76. 0.844 3.45 76. 0.832
1.05 37. 0.493 1.10 36. 0.670 1.19 33. 0.839 1.28 28. 0.832 1.31 22. 0.719
1.22 42. 0.801 1.27 41. 0.840 1.36 38. 0.868 1.45 33. 0.826 1.48 27. 0.729
1.41 48. 0.863 1.49 47. 0.879 1.58 44. 0.874 1.66 39. 0.825 1.69 34. 0.743
1.51 52. 0.873 1.62 51. 0.886 1.72 47. 0.875 1.80 43. 0.827 1.82 37. 0.751
1.63 56. 0.879 1.77 54. 0.890 1.87 51. 0.876 1.95 47. 0.830 1.97 41. 0.759
1.77 60. 0.884 1.93 59. 0.893 2.04 56. 0.878 2.12 51. 0.834 2.15 46. 0.768
1.90 64. 0.885 2.10 63. 0.895 2.23 60. 0.880 2.33 56. 0.839 2.36 51. 0.778
2.05 69. 0.885 2.30 68. 0.896 2.45 65. 0.882 2.56 61. 0.845 2.60 56. 0.788
2.17 73. 0.879 2.53 72. 0.894 2.71 69. 0.881 2.83 66. 0.847 2.87 62. 0.798
2.32 77. 0.844 3.07 77. 0.867 3.28 77. 0.856 3.39 76. 0.844 3.45 76. 0.832
1.05 37. 0.493 1.10 36. 0.670 1.19 33. 0.839 1.28 28. 0.832 1.31 22. 0.719
1.22 42. 0.801 1.27 41. 0.840 1.36 38. 0.868 1.45 33. 0.826 1.48 27. 0.729
1.41 48. 0.863 1.49 47. 0.879 1.58 44. 0.874 1.66 39. 0.825 1.69 34. 0.743
1.51 52. 0.873 1.62 51. 0.886 1.72 47. 0.875 1.80 43. 0.827 1.82 37. 0.751
1.63 56. 0.879 1.77 54. 0.890 1.87 51. 0.876 1.95 47. 0.830 1.97 41. 0.759
1.77 60. 0.884 1.93 59. 0.893 2.04 56. 0.878 2.12 51. 0.834 2.15 46. 0.768
1.90 64. 0.885 2.10 63. 0.895 2.23 60. 0.880 2.33 56. 0.839 2.36 51. 0.778
2.05 69. 0.885 2.30 68. 0.896 2.45 65. 0.882 2.56 61. 0.845 2.60 56. 0.788
2.17 73. 0.879 2.53 72. 0.894 2.71 69. 0.881 2.83 66. 0.847 2.87 62. 0.798
2.32 77. 0.844 3.07 77. 0.867 3.28 77. 0.856 3.39 76. 0.844 3.45 76. 0.832
1.05 37. 0.493 1.10 36. 0.670 1.19 33. 0.839 1.28 28. 0.832 1.31 22. 0.719
1.22 42. 0.801 1.27 41. 0.840 1.36 38. 0.868 1.45 33. 0.826 1.48 27. 0.729
1.41 48. 0.863 1.49 47. 0.879 1.58 44. 0.874 1.66 39. 0.825 1.69 34. 0.743
1.51 52. 0.873 1.62 51. 0.886 1.72 47. 0.875 1.80 43. 0.827 1.82 37. 0.751
1.63 56. 0.879 1.77 54. 0.890 1.87 51. 0.876 1.95 47. 0.830 1.97 41. 0.759
1.77 60. 0.884 1.93 59. 0.893 2.04 56. 0.878 2.12 51. 0.834 2.15 46. 0.768
1.90 64. 0.885 2.10 63. 0.895 2.23 60. 0.880 2.33 56. 0.839 2.36 51. 0.778
2.05 69. 0.885 2.30 68. 0.896 2.45 65. 0.882 2.56 61. 0.845 2.60 56. 0.788
2.17 73. 0.879 2.53 72. 0.894 2.71 69. 0.881 2.83 66. 0.847 2.87 62. 0.798
2.32 77. 0.844 3.07 77. 0.867 3.28 77. 0.856 3.39 76. 0.844 3.45 76. 0.832
1.05 37. 0.493 1.10 36. 0.670 1.19 33. 0.839 1.28 28. 0.832 1.31 22. 0.719
1.22 42. 0.801 1.27 41. 0.840 1.36 38. 0.868 1.45 33. 0.826 1.48 27. 0.729
1.41 48. 0.863 1.49 47. 0.879 1.58 44. 0.874 1.66 39. 0.825 1.69 34. 0.743
1.51 52. 0.873 1.62 51. 0.886 1.72 47. 0.875 1.80 43. 0.827 1.82 37. 0.751
1.63 56. 0.879 1.77 54. 0.890 1.87 51. 0.876 1.95 47. 0.830 1.97 41. 0.759
1.77 60. 0.884 1.93 59. 0.893 2.04 56. 0.878 2.12 51. 0.834 2.15 46. 0.768
1.90 64. 0.885 2.10 63. 0.895 2.23 60. 0.880 2.33 56. 0.839 2.36 51. 0.778
2.05 69. 0.885 2.30 68. 0.896 2.45 65. 0.882 2.56 61. 0.845 2.60 56. 0.788
2.17 73. 0.879 2.53 72. 0.894 2.71 69. 0.881 2.83 66. 0.847 2.87 62. 0.798
2.32 77. 0.844 3.07 77. 0.867 3.28 77. 0.856 3.39 76. 0.844 3.45 76. 0.832
```

```
4
0 5 10 15 20 30
0.53 0.63 0.74 0.79 0.84 0.89 0.95 1.00 1.05 1.16
1.04 293. 0.495 1.06 284. 0.681 1.13 259. 0.869 1.19 222. 0.875 1.22 179. 0.72
1.15 324. 0.829 1.20 316. 0.877 1.25 294. 0.900 1.30 261. 0.856 1.32 221. 0.74
1.28 361. 0.875 1.35 354. 0.896 1.40 335. 0.887 1.44 306. 0.839 1.46 271. 0.75
1.34 381. 0.870 1.44 375. 0.889 1.49 357. 0.876 1.53 331. 0.831 1.54 299. 0.75
1.41 402. 0.861 1.52 396. 0.881 1.58 381. 0.867 1.62 357. 0.824 1.63 329. 0.75
1.49 424. 0.853 1.61 419. 0.872 1.68 406. 0.861 1.72 387. 0.825 1.73 364. 0.76
1.56 445. 0.841 1.71 442. 0.864 1.79 432. 0.858 1.83 418. 0.832 1.85 401. 0.78
1.61 464. 0.815 1.83 462. 0.850 1.92 454. 0.848 1.97 444. 0.827 1.98 431. 0.79
1.70 478. 0.786 1.96 476. 0.825 2.06 472. 0.825 2.11 465. 0.810 2.13 457. 0.78
1.78 496. 0.635 2.10 495. 0.694 2.24 493. 0.706 2.29 489. 0.698 2.30 485. 0.68
1.04 293. 0.495 1.06 284. 0.681 1.13 259. 0.869 1.19 222. 0.875 1.22 179. 0.72
1.15 324. 0.829 1.20 316. 0.877 1.25 294. 0.900 1.30 261. 0.856 1.32 221. 0.74
1.28 361. 0.875 1.35 354. 0.896 1.40 335. 0.887 1.44 306. 0.839 1.46 271. 0.75
1.34 381. 0.870 1.44 375. 0.889 1.49 357. 0.876 1.53 331. 0.831 1.54 299. 0.75
1.41 402. 0.861 1.52 396. 0.881 1.58 381. 0.867 1.62 357. 0.824 1.63 329. 0.75
1.49 424. 0.853 1.61 419. 0.872 1.68 406. 0.861 1.72 387. 0.825 1.73 364. 0.76
1.56 445. 0.841 1.71 442. 0.864 1.79 432. 0.858 1.83 418. 0.832 1.85 401. 0.78
1.61 464. 0.815 1.83 462. 0.850 1.92 454. 0.848 1.97 444. 0.827 1.98 431. 0.79
1.70 478. 0.786 1.96 476. 0.825 2.06 472. 0.825 2.11 465. 0.810 2.13 457. 0.78
1.78 496. 0.635 2.10 495. 0.694 2.24 493. 0.706 2.29 489. 0.698 2.30 485. 0.68
0.98 277. 1.116 1.01 267. 0.192 1.09 241. 0.829 1.17 202. 0.876 1.20 156. 0.69
1.08 303. 0.718 1.12 294. 0.821 1.20 270. 0.901 1.28 234. 0.856 1.30 192. 0.70
1.20 334. 0.866 1.27 326. 0.899 1.34 304. 0.902 1.41 271. 0.842 1.43 233. 0.72
1.26 351. 0.879 1.35 344. 0.905 1.42 323. 0.897 1.48 292. 0.837 1.50 255. 0.72
1.33 370. 0.882 1.43 363. 0.904 1.51 344. 0.891 1.57 314. 0.834 1.59 279. 0.73
1.39 390. 0.879 1.52 383. 0.901 1.60 365. 0.886 1.67 338. 0.833 1.69 305. 0.74
1.46 409. 0.876 1.62 403. 0.898 1.71 387. 0.883 1.77 362. 0.834 1.79 333. 0.75
1.55 429. 0.873 1.72 424. 0.894 1.82 410. 0.883 1.89 390. 0.841 1.92 366. 0.77
1.61 447. 0.860 1.84 443. 0.888 1.96 432. 0.880 2.04 415. 0.844 2.06 396. 0.78
1.74 472. 0.812 2.09 471. 0.855 2.25 468. 0.854 2.32 463. 0.838 2.36 457. 0.81
0.95 269. 9.827 0.98 260. 0.982 1.07 233. 0.794 1.16 193. 0.876 1.20 146. 0.67
1.05 293. 0.611 1.09 284. 0.769 1.18 259. 0.898 1.26 222. 0.852 1.29 179. 0.68
1.16 321. 0.843 1.23 313. 0.891 1.31 290. 0.905 1.39 256. 0.836 1.42 215. 0.69
1.22 337. 0.872 1.30 329. 0.904 1.39 307. 0.901 1.46 274. 0.832 1.49 236. 0.70
1.28 354. 0.882 1.39 347. 0.908 1.47 326. 0.896 1.55 295. 0.829 1.57 257. 0.71
1.36 372. 0.888 1.47 365. 0.907 1.56 346. 0.892 1.64 316. 0.828 1.66 281. 0.72
1.41 391. 0.882 1.57 385. 0.905 1.66 366. 0.889 1.74 339. 0.830 1.77 306. 0.73
1.51 409. 0.884 1.67 404. 0.903 1.78 387. 0.888 1.86 363. 0.834 1.88 334. 0.74
1.57 428. 0.876 1.77 423. 0.900 1.90 410. 0.889 1.99 390. 0.845 2.02 366. 0.77
1.67 456. 0.836 2.04 454. 0.882 2.20 448. 0.877 2.29 438. 0.853 2.33 428. 0.81
0.92 262. 2.697 0.96 252. 0.238 1.05 225. 0.738 1.15 184. 0.875 1.19 136. 0.65
1.02 283. 0.382 1.06 274. 0.677 1.15 249. 0.893 1.25 211. 0.848 1.28 167. 0.66
1.13 309. 0.812 1.19 301. 0.878 1.28 277. 0.906 1.37 242. 0.830 1.40 200. 0.67
1.19 323. 0.862 1.26 315. 0.900 1.36 292. 0.903 1.45 258. 0.824 1.47 218. 0.68
1.25 339. 0.877 1.34 331. 0.908 1.44 309. 0.898 1.53 276. 0.821 1.55 237. 0.69
1.32 356. 0.887 1.43 348. 0.910 1.53 327. 0.894 1.62 296. 0.819 1.64 258. 0.69
1.39 373. 0.889 1.52 366. 0.909 1.63 346. 0.891 1.72 316. 0.820 1.74 281. 0.70
1.45 390. 0.888 1.61 384. 0.908 1.73 366. 0.890 1.83 338. 0.823 1.85 306. 0.72
1.51 408. 0.882 1.71 402. 0.907 1.85 386. 0.891 1.95 362. 0.832 1.98 333. 0.73
1.64 438. 0.857 1.97 434. 0.897 2.15 423. 0.886 2.26 406. 0.838 2.29 387. 0.77
0.87 246. 1.761 0.91 237. 1.938 1.01 209. 0.411 1.14 168. 0.873 1.18 119. 0.61
0.96 263. 0.636 1.00 254. 0.030 1.11 229. 0.871 1.23 190. 0.839 1.27 145. 0.61
1.07 285. 0.681 1.12 276. 0.821 1.23 252. 0.905 1.34 215. 0.815 1.38 172. 0.62
1.12 297. 0.801 1.18 288. 0.877 1.30 265. 0.903 1.41 229. 0.806 1.44 187. 0.63
1.18 310. 0.851 1.26 301. 0.898 1.38 278. 0.898 1.48 244. 0.800 1.52 203. 0.64
1.23 323. 0.871 1.33 315. 0.907 1.46 293. 0.893 1.57 259. 0.796 1.60 220. 0.65
1.30 337. 0.883 1.41 330. 0.909 1.55 308. 0.889 1.66 276. 0.794 1.69 238. 0.65
1.36 352. 0.885 1.50 345. 0.909 1.65 324. 0.886 1.76 294. 0.794 1.79 257. 0.66
1.42 367. 0.883 1.59 360. 0.909 1.75 341. 0.886 1.88 312. 0.797 1.90 278. 0.67
1.54 395. 0.869 1.78 390. 0.905 1.99 376. 0.894 2.14 355. 0.825 2.17 330. 0.72
```

```
-1
-1
INTAKE S1-2         D1-4              R100
 COMPRE S2-3         D5-11             R101      V5    V6
 PREMAS S3,16,4      D24-27                      V24
 ARITHY              D19-23
 COMPRE S4-5         D12-18            R102      V12
 DUCTER S16,17       D28-31            R103
 ARITHY              D32-38
 COMPRE S5-6         D39-45            R104      V39   V40
 PREMAS S6,18,7      D46-49
 PREMAS S18,19,20    D84-87
 BURNER S7-8         D50-52            R105      L1
 MIXEES S8,19,9
 TURBIN S9-10        D53-60,104,61                  V54
 MIXEES S10,20,11
 TURBIN S11-12       D62-69,108,70                  V63
 MIXFUL S12,17,13    D71-73
 DUCTER S13,14       D74-77           R106
 NOZDIV S14,15,1     D78-79           R107
 ARITHY              D120-126
 ARITHY              D127-133
 ARITHY              D134-140
 ARITHY              D141-149
 ARITHY              D150-158
 ARITHY              D159-167
 PLOTBD              D450,400,451,401,452,402
 PERFOR S1,0,0       D80-83,107,100,105,0,0,103,0,0,106,0,0
 CODEND
DATA ////
1  0.0               !              ALTITUDE
2  0.0               !INTAKE        I.S.A DEVIATION
3  0.0               !              MACH NUMBER
4  -1.0              !              PRESSURE RECOVERY

5  0.68              !              Z=(R-Rc)/(Rs-Rc)
6  0.985              !              PCN ROTATIONAL SPEED
7  1.86              !COMPRE 1       DESIGN PRESSURE RATIO
8  0.86              !              DESIGN ISENTROPIC EFFICIENCY
9  0.0               !              ERROR SELECTION
10 4.0               !               COMPRESSOR MAP NUMBER
11 0.0               !               STATOR ANGLE

12 0.72              !              Z=(R-Rc)/(Rs-Rc)
13 0.98              !               PCN ROTATIONAL SPEED
14 1.8              !COMPRE 2       DESIGN PRESSURE RATIO
15 0.89              !              DESIGN ISENTROPIC EFFICIENCY
16 1.0               !              ERROR SELECTION
17 3.0               !              COMPRESSOR MAP NUMBER
18 0.0               !              STATOR ANGLE

19 5                 !              COPY        (SAME ROTATIONAL SPEED
20 -1                !ARITHY        B.D.         FOR COMPRESSOR 1 ,2)
21 13.0              !              B.D. ITEM NUMBER
22 -1                !              B.D.
23 6.0               !              B.D. ITEM NUMBER

24 0.585              !               BYPASS RATIO
25 0.0               !PREMAS        LOSS IN W          ( BYPASS)
26 1.0               !              PRESSURE RATIO
27 0.01              !              LOSS IN PRESSURE

28 0.0               !              SWITCH (NO REHEATING REQUIRED)
```

```
29 0.03                   !DUCTER         PRESSURE LOSS/INLET TOTAL PRESSURE
30 0.0                    !               COMBUSTION EFFICIENCY
31 1000000.0              !               LIMITING VALUE OF FUEL

32 1                      !               ADDITION
33 -1                     !               BD.          {WC = WC1 + WC2 }
34 108                    !ARITHY         BD. ITEM NUMBER
35 -1                     !               BD.
36 101                    !               BD. ITEM NUMBER
37 -1                     !               BD.
38 102                    !               BD. ITEM NUMBER

39 0.69                    !               Z=(R-Rc)/(Rs-Rc)
40 0.98                    !               PCN ROTATIONAL SPEED
41 2.54                    !COMPRE 3       DESIGN PRESSURE RATIO
42 0.88                   !               DESIGN ISENTROPIC EFFICIENCY
43 1.0                    !               ERROR SELECTION
44 2.0                    !               COMPRESSOR MAP NUMBER
45 0.0                    !               STATOR ANGLE

46 0.15                   !               BYPASS RATIO
47 0.0                    !PREMAS         LOSS IN W          ( COOLING )
48 1.0                    !               PRESSURE RATIO
49 0.0                    !               LOSS IN PRESSURE

50 0.05                   !               TOTAL PRESSURE LOSS
51 0.99                    !BURNER         COMBUSTION
52 -1.0                   !               FUEL FLOW

53 100000.0               !               AUXILIARY WORK
54 -1.0                   !          RELATIVE NO-DIMENSIONAL INLET MASS FLOW
55 -1.0                   !TURBIN 1       RELATIVE NON-DIMENSIONAL SPEED CN
56 0.9                    !               DESIGN ISENTROPIC EFFICIENCY
57 -1.0                   !               RELATIVE ROTATIONAL SPEED
58 3.0                    !               COMPRESSOR NUMBER
59 4.0                    !               TURBINE MAP NUMBER
60 -1.0                   !               POWER LAW INDEX
61 0.0                    !               NGV ANGLE

62 0.0                    !               AUXILIARY WORK
63 -1.0                   !          RELATIVE NO-DIMENSIONAL INLET MASS FLOW
64 0.6                     !               RELATIVE NON-DIMENSIONAL SPEED CN
65 0.9                    !TURBIN 2       DESIGN ISENTROPIC EFFICIENCY
66 -1.0                   !               RELATIVE ROTATIONAL SPEED
67 1.0                    !               COMPRESSOR NUMBER
68 4.0                    !               TURBINE MAP NUMBER
69 -1.0                   !               POWER LAW INDEX
70 0.0                    !               NGV  ANGLE

71 1.0                    !          NUMBER OF COMPRESSOR PROVIDING STREAM 2
72 1.0                    !MIXFUL         MACH NUMBER IN BD(3)
73 0.42                   !               MACH NUMBER

74 1.0                    !               SWITCH (NO REHEATING REQUIRED)
75 0.03                   !DUCTER         PRESSURE LOSS/INLET TOTAL PRESSURE
76 0.91                   !               COMBUSTION EFFICIENCY
77 1000000.0              !               LIMITING VALUE OF FUEL

78 -1.0                   !NOZDIV         FOR AREAS
79 -1.0                   !               THROAT AREA

80 -1.0                   !               POWER FOR POWER TURBINE
81 -1.0                   !PERFOR         PROPELLER EFFICIENCY
82 0.0                    !               SCALING INDEX
83 182000                 !               REQUIRED DESIGN-POINT NET THRUST

84 0.6                    !               BYPASS RATIO
85 0.0                    !PREMAS         LOSS IN W          ( COOLING )
86 1.0                    !               PRESSURE RATIO
```

```
87 0.0                     !                LOSS IN PRESSURE

120 4
121 -1
122 400                    !NDMF LPC
123 3
124 4
125 2
126 4

127 4
128 -1
129 401
130 5
131 4                     !NDMF IPC
132 4
133 4

134 4
135 -1
136 402
137 6
138 4
139 5
140 4                     !NDMF HPC

141 29
142 -1
143 450
144 3
145 2
146 2
147 6
148 2
149 4

150 29
151 -1
152 451
153 4
154 2
155 4
156 6
157 4
158 4

159 29
160 -1
161 452
162 5
163 2
164 5
165 6
166 5
167 4
-1
1 2 453.277                  !INTAKE MASS FLOW
8 6 1240.0                   !TET TURBINE ENTRY TEMPRATURE
-1
78 -2
1 460
3 0.5
-1
8 6 1440
-1
1 2290
3 0.6
-1
8 6 1350
```

```
-1
1 9150
3 0.95
70 -2
79 1.54
-1
8 6 1160
17 8 0.85
-1
-3
11 5
18 -5
79 1.42
-1
17 8 1
12 8 1
8 6 1250
-1
11 10
18 -6
78 1
74 2
9 1
-1
17 8 0.6
12 8 1.0
8 6 1210
14 6 1020
-1
11 15
1 9750
3 1.05
-1
8 6 1250
14 6 1120
-1
1 12250
3 1.6
11 20
18 -5
-1
8 6 1350
14 6 1400
-1
78 -1
9 0
74 1
11 25
1 18750
3 2.7
79 1.04
-1
8 6 1750
12 8 0.97
17 8 0.49
15 8 3.0
-1
74 2
11 30
18 -2
 -1
 14 6 1250
 8 6 1800
 -1
74 1
 -1
17 8 0.45
 -1
-3
```

TURBOMATCH OUTPUT FILE

Turbofan-Turbojet Engine

```
TURBOMATCH SCHEME-VAX VERSION (11/11/94)

 LIMITS:100 Codewords, 800 Brick Data Items, 50 Station Vector
 15 BD Items printable by any call of:-
 OUTPUT, OUTPBD, OUTPSV, PLOTIT, PLOTBD or PLOTSV

***** DESIGN POINT ENGINE CALCULATIONS *****


 ***** AMBIENT AND INLET PARAMETERS *****
 Alt. =       0.0       I.S.A. Dev. =     0.000       Mach No. =  0.00
 Etar = 1.0000          Momentum Drag =       0.00

 ***** COMPRESSOR  1 PARAMETERS *****
 PRSF =  0.10358E+01      ETASF =  0.10086E+01       WASF =  0.10022E+01
 Z = 0.68000              PR =   1.860               ETA = 0.86000
 PCN =   0.9850           CN = 0.98500               COMWK =  0.29598E+08
 STATOR ANGLE =    0.00

 ***** COMPRESSOR  2 PARAMETERS *****
 PRSF =  0.99651E+00      ETASF =  0.99739E+00       WASF =  0.98846E+00
 Z = 0.72000              PR =   1.800               ETA = 0.89000
 PCN =   0.9850           CN = 0.98500               COMWK =  0.13703E+08
 STATOR ANGLE =    0.00

 ***** COMPRESSOR  3 PARAMETERS *****
 PRSF =  0.98491E+00      ETASF =  0.99476E+00       WASF =  0.10016E+01
 Z = 0.69000              PR =   2.540               ETA = 0.88000
 PCN =   0.9800           CN = 0.98000               COMWK =  0.27786E+08
 STATOR ANGLE =    0.00

 ***** COMBUSTION CHAMBER PARAMETERS *****
 ETASF =  0.99000E+00
 ETA = 0.99000            DLP = 0.4252               WFB =   3.0201

 ***** TURBINE  1 PARAMETERS *****
 CNSF =  0.72259E+02      ETASF =  0.10567E+01       TFSF =  0.54146E+00
 DHSF =  0.76730E+04
 TF =  414.346             ETA = 0.90000              CN =  2.060
 AUXWK =  0.10000E+06      NGV ANGLE = 0.00

 ***** TURBINE  2 PARAMETERS *****
 CNSF =  0.66968E+02      ETASF =  0.10567E+01       TFSF =  0.31821E+00
 DHSF =  0.12920E+05
 TF =  414.346             ETA = 0.90000              CN =  2.060
 AUXWK =  0.00000E+00      NGV ANGLE = 0.00

 ***** MIXING MACH NUMBERS *****
 Station 12, M= 0.421     Station 17,M = 0.429    Station 13,M = 0.444

 ***** DUCT/AFTER BURNING 1 PARAMETERS *****
 ETA =  0.9100            DLP = 0.0537              WFB =   0.0000

 ***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
 Throat Area =  1.5193    Throat Velocity =  432.63    Throat Mach No. =  1.0000
 Exit Area =  1.5261      Exit Velocity =  406.23      Exit Mach No. =  0.9287
 Nozzle Coeff. =  0.9794  Gross Thrust = 181544.36

 Nozzle exit and throat areas are fixed.
 AREA RATIO (EXIT/THROAT): 1.00
 Scale Factor on above Mass Flows, Areas, Thrusts & Powers =     1.0000
```

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	453.277	1.00000	1.00000	288.15	288.15	0.0	******
2	0.00000	453.277	******	1.00000	******	288.15	******	******
3	0.00000	453.277	******	1.86000	******	353.11	******	******
4	0.00000	188.110	******	1.86000	******	353.11	******	******
5	0.00000	188.110	******	3.34800	******	425.10	******	******
6	0.00000	188.110	******	8.50392	******	568.66	******	******
7	0.00000	159.893	******	8.50392	******	568.66	******	******
8	0.01889	162.914	******	8.07872	******	1240.00	******	******
9	0.01708	179.843	******	8.07872	******	1181.67	******	******
10	0.01708	179.843	******	4.70015	******	1051.97	******	******
11	0.01605	191.130	******	4.70015	******	1025.35	******	******
12	0.01605	191.130	1.59909	1.79763	804.95	829.25	234.3	1.1633
13	0.00666	456.297	1.56810	1.79074	540.83	560.85	205.3	2.1713
14	0.00666	456.297	******	1.73701	******	560.85	******	******
15	0.00666	456.297	1.00000	1.73701	481.68	560.85	406.2	1.5261
16	0.00000	265.167	******	1.87000	******	353.11	******	******
17	0.00000	265.167	1.59909	1.81390	340.60	353.11	158.7	1.0081
18	0.00000	28.216	******	8.50392	******	568.66	******	******
19	0.00000	16.930	******	8.50392	******	568.66	******	******
20	0.00000	11.287	******	8.50392	******	568.66	******	******

```
  Gross Thrust = 181544.36
Momentum Drag =      0.00
   Net Thrust = 181544.36
    Fuel Flow =    3.0201
       s.f.c. =  16.63537
   Sp. Thrust =   400.515
         Time Now 16:41:32

***********************************************
  1 9150
  3 0.95
  -1
  8 6 1160
  -1
                                              Time Now 16:41:32

***********************************************

     ***** OFF DESIGN ENGINE CALCULATIONS. Converged after   5 Loops *****


***** AMBIENT AND INLET PARAMETERS *****
Alt. =   9150.0       I.S.A. Dev. =    0.000       Mach No. =  0.95
Etar = 1.0000         Momentum Drag =  71264.70

***** COMPRESSOR  1 PARAMETERS *****
PRSF =  0.10358E+01       ETASF =  0.10086E+01      WASF =  0.10022E+01
Z = 0.64500               PR =   1.834              ETA = 0.86197
PCN =   0.9473            CN = 0.97848              COMWK =  0.14729E+08
STATOR ANGLE =    0.00

***** COMPRESSOR  2 PARAMETERS *****
PRSF =  0.99651E+00       ETASF =  0.99739E+00      WASF =  0.98846E+00
Z = 0.73313               PR =   1.795              ETA = 0.89025
PCN =   0.9473            CN = 0.98081              COMWK =  0.68436E+07
STATOR ANGLE =    0.00

***** COMPRESSOR  3 PARAMETERS *****
PRSF =  0.98491E+00       ETASF =  0.99476E+00      WASF =  0.10016E+01
Z = 0.69335               PR =   2.530              ETA = 0.87977
PCN =   0.9438            CN = 0.97732              COMWK =  0.13887E+08
STATOR ANGLE =    0.00

***** COMBUSTION CHAMBER PARAMETERS *****
ETA = 0.99000             DLP = 0.2209              WFB =   1.4969

***** TURBINE  1 PARAMETERS *****
CNSF =  0.72259E+02       ETASF =  0.10567E+01      TFSF =  0.54146E+00
DHSF =  0.76730E+04
TF =  414.255              ETA = 0.89954             CN =  2.051
AUXWK =  0.10000E+06       NGV ANGLE = 0.00

***** TURBINE  2 PARAMETERS *****
CNSF =  0.66968E+02       ETASF =  0.10567E+01      TFSF =  0.31821E+00
DHSF =  0.12920E+05
TF =  413.853              ETA = 0.89966             CN =  2.050
AUXWK =  0.00000E+00       NGV ANGLE = 0.00

***** MIXING MACH NUMBERS *****
Station 12, M= 0.416    Station 17,M = 0.436    Station 13,M = 0.444

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA =  0.9100             DLP = 0.0282              WFB =   0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
Throat Area =  1.5193     Throat Velocity =  416.45    Throat Mach No. =  1.0000
Exit Area =  1.6750       Exit Velocity =  535.91      Exit Mach No. =  1.3758
Nozzle Coeff. =  0.9793   Gross Thrust = 130602.93
```

Nozzle exit area floating/throat area fixed.
AREA RATIO (EXIT/THROAT): 1.10

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	247.345	0.29663	0.53043	228.68	270.10	288.1	******
2	0.00000	247.345	******	0.53043	******	270.10	******	******
3	0.00000	247.345	******	0.97267	******	329.41	******	******
4	0.00000	101.188	******	0.97267	******	329.41	******	******
5	0.00000	101.183	******	1.74632	******	396.45	******	******
6	0.00000	101.183	******	4.41870	******	530.61	******	******
7	0.00000	86.006	******	4.41870	******	530.61	******	******
8	0.01741	87.503	******	4.19779	******	1160.00	******	******
9	0.01574	96.609	******	4.19779	******	1105.11	******	******
10	0.01574	96.609	******	2.44259	******	982.35	******	******
11	0.01479	102.680	******	2.44259	******	957.36	******	******
12	0.01479	102.680	0.83799	0.93903	750.53	772.87	224.1	1.1633
13	0.00605	248.840	0.82177	0.93867	500.48	519.22	197.7	2.1713
14	0.00605	248.840	******	0.91051	******	519.22	******	******
15	0.00605	248.840	0.29663	0.91051	379.55	519.22	535.9	1.6750
16	0.00000	146.160	******	0.98267	******	329.41	******	******
17	0.00000	146.160	0.83624	0.95319	317.33	329.41	155.8	1.0081
18	0.00000	15.177	******	4.41870	******	530.61	******	******
19	0.00000	9.106	******	4.41870	******	530.61	******	******
20	0.00000	6.071	******	4.41870	******	530.61	******	******

Gross Thrust = 130602.93
Momentum Drag = 71264.70
Net Thrust = 59338.23
Fuel Flow = 1.4969
s.f.c. = 25.22725
Sp. Thrust = 239.901
****
1 18750
3 2.7
-1
8 6 1800
-1

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 6 Loops *****

***** AMBIENT AND INLET PARAMETERS *****
Alt. = 18750.0   I.S.A. Dev. = 0.000   Mach No. = 2.70
Etar = 0.8465   Momentum Drag = 213895.69

***** VARIABLE COMPRESSOR 1 PARAMETERS *****
PRSF = 0.10358E+01   ETASF = 0.10086E+01   WASF = 0.10022E+01
Z = 0.89374   PR = 1.671   ETA = 0.81082
PCN = 1.2882   CN = 0.94958   COMWK = 0.27789E+08
STATOR ANGLE = 30.00

***** VARIABLE COMPRESSOR 2 PARAMETERS *****
PRSF = 0.99651E+00   ETASF = 0.99739E+00   WASF = 0.98846E+00
Z = 0.92325   PR = 1.828   ETA = 0.86726
PCN = 1.2882   CN = 0.96496   COMWK = 0.21198E+08
STATOR ANGLE = -2.00

***** COMPRESSOR 3 PARAMETERS *****
PRSF = 0.98491E+00   ETASF = 0.99476E+00   WASF = 0.10016E+01
Z = 0.53470   PR = 2.177   ETA = 0.88557
PCN = 1.2171   CN = 0.91241   COMWK = 0.32844E+08
STATOR ANGLE = 0.00

***** COMBUSTION CHAMBER PARAMETERS *****
ETA = 0.99000   DLP = 0.4745   WFB = 3.5576

***** TURBINE 1 PARAMETERS *****
CNSF = 0.72259E+02   ETASF = 0.10567E+01   TFSF = 0.54146E+00
DHSF = 0.76730E+04
TF = 411.626   ETA = 0.90424   CN = 2.117
AUXWK = 0.10000E+06   NGV ANGLE = 0.00

***** TURBINE 2 PARAMETERS *****
CNSF = 0.66968E+02   ETASF = 0.10567E+01   TFSF = 0.31821E+00
DHSF = 0.12920E+05
TF = 406.187   ETA = 0.90998   CN = 2.212
AUXWK = 0.00000E+00   NGV ANGLE = 0.00

***** MIXING MACH NUMBERS *****
Station 12, M= 0.469   Station 17,M = 0.490   Station 13,M = 0.494

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA = 0.9100   DLP = 0.0620   WFB = 0.0000

ARATIO = 2.523   Out of Range: use Data for 2.150
***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
Throat Area = 1.0700   Throat Velocity = 579.48   Throat Mach No. = 1.0000
Exit Area = 2.7000   Exit Velocity = 1077.69   Exit Mach No. = 2.4308
Nozzle Coeff. = 0.9526   Gross Thrust = 296421.00

Nozzle exit and throat areas are fixed.

AREA RATIO (EXIT/THROAT): 2.52
Nozzle is Underexpanded.

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	268.363	0.06577	1.53692	216.65	530.28	797.0	******
2	0.00000	268.363	******	1.30097	******	530.28	******	******
3	0.00000	268.363	******	2.17411	******	629.28	******	******
4	0.00000	155.412	******	2.17411	******	629.28	******	******
5	0.00000	155.337	******	3.97429	******	756.40	******	******
6	0.00000	155.337	******	8.65235	******	946.85	******	******
7	0.00000	132.037	******	8.65235	******	946.85	******	******
8	0.02694	135.594	******	8.17785	******	1800.00	******	******
9	0.02436	149.575	******	8.17785	******	1726.47	******	******
10	0.02436	149.575	******	4.85734	******	1554.67	******	******
11	0.02290	158.895	******	4.85734	******	1521.52	******	******
12	0.02290	158.895	1.79559	2.06738	1233.77	1275.01	318.7	0.9700
13	0.01323	272.499	1.76600	2.06808	981.53	1020.43	302.1	1.4200
14	0.01323	272.499	******	2.00604	******	1020.43	******	******
15	0.01323	272.499	0.12671	2.00604	496.86	1020.43	1077.7	2.7000
16	0.00000	113.604	******	2.18411	******	629.28	******	******
17	0.00000	113.604	1.80248	2.11859	602.19	629.28	238.8	0.4500
18	0.00000	23.301	******	8.65235	******	946.85	******	******
19	0.00000	13.980	******	8.65235	******	946.85	******	******
20	0.00000	9.320	******	8.65235	******	946.85	******	******

Gross Thrust = 296421.00
Momentum Drag = 213895.69
Net Thrust = 82525.30
Fuel Flow = 3.5576
s.f.c. = 43.10938
Sp. Thrust = 307.513
Time Now 16:41:33

INPUT FILE FOR THE L.P. COMPRESSOR

For Compdes Program

Turbofan-Turbojet Engine.
Design point is The Take-Off Point.

```
L.P COMPRESSOR DESIGN         !TITLE
2                             !NOMBER OF STAGE
3                             !ANNULUS GEOMETRY SELECTOR
1                             !IGV's SELECTION
0.4                           !HUB/TIP RATIO
88                            !ISENTROPIC EFFICIENCY
200 200 200 200               !INLET AXIAL VELOCITY
30 33.48                      !TEMP. RISE PER STAGE
288.15                        !INLET TOTAL TEMPERATURE
1                             !INLET TOTAL PRESSURE
1.86                          !DESIGN PRESSURE RATIO
453.3                         !INLET MASS FLOW
1                             !VORTEX DESIGN SELECTOR
0 0 0 0                       !STAGE INLET SWIRL ANGLE
4300                          !ROTATIONAL SPEED
0.98 0.93                     !STAGE WORK DONE FACTORS
0.8 0.7 0.8 0.7               !S/C RATIO
3 2.75                        !ROTOR H/C RATIO
0.8 0.7                       !BMH  REACTION
0.01 0.01                     !ROTOR TIP CLEARANCE
0.00001                       !EQUIVALENT SAND ROUGHNESS
```

INPUT FILE FOR THE I.P. COMPRESSOR

For Compdes Program

Turbofan-Turbojet Engine.
Design point is The Take-Off Point.

```
I.P COMPRESSOR DESIGN        !TITLE
3                            !NOMBER OF STAGE
1                            !ANNULUS GEOMETRY SELECTOR
1                            !IGV's SELECTION
0.77                         !HUB/TIP RATIO
88                           !ISENTROPIC EFFICIENCY
170 180 180 199 180          !INLET AXIAL VELOCITY
22 24 26.52                  !TEMP. RISE PER STAGE
351.5                        !INLET TOTAL TEMPERATURE
1.86                         !INLET TOTAL PRESSURE
1.8                          !DESIGN PRESSURE RATIO
188.11                       !INLET MASS FLOW
1                            !VORTEX DESIGN SELECTOR
0 0 0 0 0                    !STAGE INLET SWIRL ANGLE
4000                         !ROTATIONAL SPEED
0.98 0.95 0.92               !STAGE WORK DONE FACTORS
0.8 0.7 0.8 0.7 0.8 0.7      !S/C RATIO
3.5 3.2 3                    !ROTOR H/C RATIO
0.8 0.7 0.7                  !BMH  REACTION
0.01 0.01 0.01               !ROTOR TIP CLEARANCE
0.00001                      !EQUIVALENT SAND ROUGHNESS
```

INPUT FILE FOR THE H.P. COMPRESSOR

For Compdes Program

Turbofan-Turbojet Engine.
Design point is The Take-Off Point.

```
H.P COMPRESSOR DESIGN          !TITLE
3                              !NOMBER OF STAGE
3                              !ANNULUS GEOMETRY SELECTOR
0                              !IGV's SELECTION
0.72                           !HUB/TIP RATIO
88                             !ISENTROPIC EFFICIENCY
200 200 200 200 200            !INLET AXIAL VELOCITY
45.27 50 48                    !TEMP. RISE PER STAGE
424.15                         !INLET TOTAL TEMPERATURE
3.348                          !INLET TOTAL PRESSURE
2.54                           !DESIGN PRESSURE RATIO
188.11                         !INLET MASS FLOW
1                              !VORTEX DESIGN SELECTOR
0 15 15 10 0                   !STAGE INLET SWIRL ANGLE
8200                           !ROTATIONAL SPEED
0.99 0.95 0.92                 !STAGE WORK DONE FACTORS
0.8 0.75 0.8 0.75 0.8 0.75     !S/C RATIO
2 2 2                          !ROTOR H/C RATIO
0.82 0.71 0.69                 !BMH REACTION
0.01 0.01 0.01                 !ROTOR TIP CLEARANCE
0.00001                        !EQUIVALENT SAND ROUGHNESS
```

INPUT FILE FOR THE L.P. COMPRESSOR

From Compdes To Howell Program

Turbofan-Turbojet Engine
Off-Design Performance
The Effect Of Variable Stator Angle

```
L.P COMPRESSOR DESIGN          !TITLE
    0        2        1  101.0000     288.0000
 4300.000          1
  0.70   0.00   1.00   0.00
 48.11242      41.16166      24.02962      -5.515311
 915.5967      915.5967      915.5967      915.5967
 366.2388      409.6076      428.2558      480.2817
   0.80    0.07    0.50    0.00    29.51
   0.70    0.06    0.50    0.00    42.54
  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  1.026  1.010  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  1.000
 51.06388      46.18544      25.13906      -5.798115
 915.5967      915.5967      915.5967      915.5967
 527.0905      558.1517      572.6741      614.1480
   0.80    0.07    0.50    0.00    42.22
   0.70    0.06    0.50    0.00    57.81
  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  1.026  1.010  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  1.000
    10
 4728.769         -5
 4298.881         -5
 4083.937         -5
 3868.993         -5
 3654.049         -5
 3439.105         -5
 3224.161         -5
 3009.217         -5
 2579.329         -5
 2149.440         -5
    0
```

INPUT FILE FOR THE I.P. COMPRESSOR

From Compdes To Howell Program

Turbofan-Turbojet Engine
Off-Design Performance
The Effect Of Variable Stator Angle

```
I.P COMPRESSOR DESIGN          !TITLE
   0       3       1  101.0000    288.0000
 4000.000          1
  0.70   0.00   1.00   0.00
 49.70322     43.97427     21.43363     -4.867877
 710.7208     708.2992     707.2291      703.8853
 552.1349     552.1349     552.1349      552.1349
   0.80    0.07    0.50    0.00   111.18
   0.70    0.06    0.50    0.00   134.04
  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  1.026  1.010  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  1.000
 49.92091     40.15902     24.56240     -5.650634
 692.4907     690.1075     689.0831      686.0092
 552.1349     552.1349     552.1349      552.1349
   0.80    0.07    0.50    0.00   114.36
   0.70    0.06    0.50    0.00   142.63
  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  1.026  1.010  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  1.000
 46.89745     32.24291     25.90707     -5.996238
 667.4434     665.1152     664.2004      661.2731
 552.1349     552.1349     552.1349      552.1349
   0.80    0.07    0.50    0.00   125.75
   0.70    0.06    0.50    0.00   147.89
  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  1.026  1.010  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  1.000
    10
 3982.780        -5
 3620.708        -5
 3439.673        -5
 3258.637        -5
 3077.602        -5
 2896.567        -5
 2715.531        -5
 2534.496        -5
 2172.425        -5
 1810.354        -5
    0
```

INPUT FILE FOR THE H.P. COMPRESSOR

From Compdes To Howell Program

Turbofan-Turbojet Engine
Off-Design Performance
The Effect Of Variable Stator Angle

```
H.P COMPRESSOR DESIGN     !TITLE
   0      3      1  101.0000    288.0000
 8200.000        1
  0.70   0.00   1.00   0.00
 51.15043     41.58833     37.93142     8.611261
 501.6899     501.6899     501.6899     501.6899
 359.7151     369.1215     373.3503     384.6435
   0.80   0.07   0.50   0.00   49.93
   0.75   0.06   0.50   0.00   62.22
  0.000  0.000  0.000  0.000  0.000  0.000
 15.000  1.026  1.010  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  1.000
 52.51182     40.97237     40.17252     1.634626
 501.6899     501.6899     501.6899     501.6899
 393.7060     400.7422     403.9630     412.6179
   0.80   0.07   0.50   0.00   68.01
   0.75   0.06   0.50   0.00   84.04
  0.000  0.000  0.000  0.000  0.000  0.000
 15.000  1.026  1.010  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  1.000
 54.37367     46.51597     36.72955     -9.460836
 501.6899     501.6899     501.6899     501.6899
 419.0786     424.1065     426.5900     433.7091
   0.80   0.07   0.50   0.00   90.53
   0.75   0.06   0.50   0.00  109.12
  0.000  0.000  0.000  0.000  0.000  0.000
 10.000  1.026  1.010  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000  0.000
  0.000  0.000  0.000  0.000  0.000  1.000
    10
 7432.640        -5
 6756.945        -5
 6419.098        -5
 6081.250        -5
 5743.403        -5
 5405.556        -5
 5067.708        -5
 4729.861        -5
 4054.167        -5
 3378.472        -5
```

OUTPUT FILE FOR THE L.P. COMPRESSOR

From Compdes Program

Turbofan-Turbojet Engine

PROPERTY DISTRIBUTION THROUGH A 2 STAGE AXIAL COMPRESSOR WITH IGVs

CONSTANT OUTER RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)----------------453.30
OVERALL PRESSURE RATIO---------- 1.86
INLET AXIAL VELOCITY (m/s)-----200.00
ISENTROPIC EFFICIENCY (%)-------88.00
OVERALL TEMPERATURE RISE (K)--- 63.48
INLET TOTAL TEPERATURE (K)---288.15
INLET TOTAL PRESSURE (atm)--- 1.00
ROTATIONAL SPEED (rpm)------- 4300.00
INLET IGV'S STA 1 STA 2 STA 3 STA 4 STA 5 STA 6 STA 7 STA 8 STA 9
TOTAL TEMPERATURE RISE (K)
30.00 33.48 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL TEMPERATURE (K)
288.15 288.15 318.15 351.63 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AVERAGE TOTAL TEMPERATURE (K)
303.15 334.89 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET STATIC TEMPERATURE (K)
268.21 268.21 298.24 331.78 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL PRESSURE (atm)
1.000 1.000 1.363 1.866 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET STATIC PRESSURE (atm)
0.778 0.778 1.087 1.522 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET DENSITY (kg/cubic m)
1.0245 1.0245 1.2872 1.6191 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET SPECIFIC HEAT (kJ/kg K)
1.0030 1.0030 1.0048 1.0075 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

AVERAGE SPECIFIC HEAT (kJ/kg K)
1.0038 1.0060 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET ANNULUS AREA (square m)
2.2123 2.2123 1.7608 1.3998 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET INNER RADIUS (cm)
36.62 36.62 52.71 62.67 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET OUTER RADIUS (cm)

91.56 91.56 91.56 91.56 0.00 0.00 0.00 0.00 0.00 0.00 0.00

BLADE HEIGHT (cm)
54.94 54.94 38.85 28.89 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET HUB/TIP RATIO
0.400 0.400 0.576 0.684 0.000 0.000 0.000 0.000 0.000 0.000 0.000

ABSOLUTE OUTLET VELOCITY (m/s)
200.00 200.00 200.00 200.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET SWIRL ANGLE (DEGREES)
0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AXIAL VELOCITY (m/s)
200.00 200.00 200.00 200.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

PRESSURE RATIO
1.3633 1.3686 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

ENTHALPY RISE (kJ/kg)
30.114 33.681 0.000 0.000 0.000 0.000 0.000 0.000 0.000

COMPRESSOR POWER (kW)--------- 28918.406

OVERALL PRESSURE RATIO-------- 1.8659

L.P COMPRESSOR DESIGN ITITLE

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT OUTER RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9
OUTSIDE RADIUS	91.56	91.56	91.56	0.00	0.00	0.00	0.00	0.00	0.00	0.00
HUB RADIUS	36.62	52.71	62.67	0.00	0.00	0.00	0.00	0.00	0.00	0.00
INLET HEIGHT	54.936	38.851	28.890	0.000	0.000	0.000	0.000	0.000	0.000	0.000
INLET HUB/TIP RATIO	0.4000	0.5757	0.6845	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
ASPECT RATIO	3.0000	2.7500	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
MID CHORD HEIGHT-ROTOR	52.248	37.298	0.000	0.000	0.000	0.000	0.000	0.000	0.000
MID CHORD HEIGHT-STATOR	44.023	32.219	0.000	0.000	0.000	0.000	0.000	0.000	0.000
TRUE CHORD-ROTOR	17.416	13.563	0.000	0.000	0.000	0.000	0.000	0.000	0.000
TRUE CHORD-STATOR	14.674	11.716	0.000	0.000	0.000	0.000	0.000	0.000	0.000
ROTOR AXIAL CHORD	12.151	8.703	0.000	0.000	0.000	0.000	0.000	0.000	0.000
ROTOR AXIAL SPACE	5.225	4.069	0.000	0.000	0.000	0.000	0.000	0.000	0.000
STATOR AXIAL CHORD,	14.576	11.620	0.000	0.000	0.000	0.000	0.000	0.000	0.000
STATOR AXIAL SPACE	4.402	3.515	0.000	0.000	0.000	0.000	0.000	0.000	0.000
TOTAL STAGE LENGTH	36.354	27.906	0.000	0.000	0.000	0.000	0.000	0.000	0.000
NUMBER OF ROTOR BLADES	29.5	42.2	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NUMBER OF STATOR BLADES	42.5	57.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0

TOTAL COMPRESSOR LENGTH 64.260

STAGE 1

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	171.0	229.8	288.6	347.4	406.2	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	200.0	200.0	200.0	200.0	200.0	M/S
REL. ROTOR INLET VELOCITY	(V1)	263.1	304.6	351.1	400.9	452.8	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	200.2	221.9	270.5	327.2	386.4	M/S
ABS. STATOR INLET VELOCITY	(V3)	268.9	240.6	226.6	218.7	213.8	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	200.0	200.0	200.0	200.0	200.0	M/S

AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	0.655	0.573	0.486	0.401	0.324	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	0.881	0.962	1.001	1.022	1.035	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.80	0.93	1.07	1.22	1.38	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.80	0.71	0.66	0.64	0.62	
ROTOR DECELERATION (V2/V1)	0.76	0.73	0.77	0.82	0.85	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.42	0.47	0.41	0.33	0.27	
STATOR DECELERATION (V0/V3)	0.74	0.83	0.88	0.91	0.94	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	1.05	0.58	0.37	0.25	0.19	
FLOW COEFFICENT (VA/U)	1.17	0.87	0.69	0.58	0.49	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	0.0	0.0	0.0	0.0	0.0	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	40.5	49.0	55.3	60.1	63.8	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	-2.5	25.7	42.3	52.3	58.8	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	41.9	33.8	28.0	23.9	20.7	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION	43.0	23.3	13.0	7.8	5.0	DEGREES
STATOR DEFLECTION	41.9	33.8	28.0	23.9	20.7	DEGREES
REACTION	47.4	70.9	81.6	87.3	90.7	PERCENT
SPACE CHORD RATIO (ROTOR)	0.658	0.760	0.800	0.815	0.821	
SPACE CHORD RATIO (STATOR)	0.443	0.574	0.700	0.824	0.946	
DIFFUSION FACTOR (ROTOR)	0.464	0.439	0.351	0.274	0.215	
DIFFUSION FACTOR (STATOR)	0.404	0.328	0.282	0.252	0.232	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.746	1.721	1.563	1.439	1.356	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.687	1.542	1.469	1.428	1.402	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00428	0.03563	0.05747	0.10358	
SHOCK LOSS COEFFICIENT (STATOR)	0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.22E+06	.23E+06	.24E+06	.26E+06	.27E+06	
REYNLDS NUMBER (STATOR)	.29E+06	.30E+06	.31E+06	.32E+06	.33E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.02375	0.01927	0.02127	0.02466	0.02859
PROFILE LOSS COEFFICIENT (STATOR)	0.03296	0.02789	0.02404	0.02078	0.01840
SECONDARY LOSS COEFFICIENT (ROTOR)			0.00818		
SECONDARY LOSS COEFFICIENT (STATOR)			0.01152		
ANNULUS LOSS COEFFICIENT (ROTOR)			0.00797		
ANNULUS LOSS COEFFICIENT (STATOR)			0.00576		
TOTAL LOSS COEFFICIENT (ROTOR)			0.07984		
TOTAL LOSS COEFFICIENT (STATOR)			0.04209		

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.012

ROTOR ISENTROPIC EFFICIENCY 87.38 PERCENT

STAGE ISENTROPIC EFFICIENCY 84.15 PERCENT

STAGE PRESSURE RATIO 1.34

STAGE 2

	BLADE ROOT	25% HEIGHT	BLADE MID	75%	BLADE TIP	
BLADE SPEED (U)	240.8	282.8	324.8	366.8	408.8	M/S
ABS. STATOR OUTLET VELOCITY (V0)	200.0	200.0	200.0	200.0	200.0	M/S
REL. ROTOR INLET VELOCITY (V1)	313.1	346.4	381.5	417.8	455.1	M/S
REL. ROTOR OUTLET VELOCITY (V2)	219.5	252.9	292.4	334.5	377.5	M/S
ABS. STATOR INLET VELOCITY (V3)	250.2	237.5	229.0	223.0	218.7	M/S
ABS. STATOR OUTLET VELOCITY (V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S

STATIC PRESSURE AT ROTOR INLET (PS1)	0.789	0.708	0.627	0.548	0.475	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	1.303	1.347	1.376	1.396	1.410	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.90	1.00	1.10	1.21	1.31	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.70	0.66	0.63	0.62	0.60	
ROTOR DECELERATION (V2/V1)	0.70	0.73	0.77	0.80	0.83	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.51	0.47	0.41	0.36	0.31	
STATOR DECELERATION (V0/V3)	0.80	0.84	0.87	0.90	0.91	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	0.62	0.45	0.34	0.27	0.22	
FLOW COEFFICENT (VA/U)	0.83	0.71	0.62	0.55	0.49	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	0.0	0.0	0.0	0.0	0.0	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	50.3	54.7	58.4	61.4	63.9	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	24.3	37.7	46.8	53.3	58.0	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	36.9	32.6	29.1	26.3	23.9	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION	26.0	17.0	11.5	8.1	5.9	DEGREES
STATOR DEFLECTION	36.9	32.6	29.1	26.3	23.9	DEGREES
REACTION	68.8	77.4	82.8	86.5	89.2	PERCENT
SPACE CHORD RATIO (ROTOR)	0.765	0.789	0.800	0.804	0.805	
SPACE CHORD RATIO (STATOR)	0.535	0.618	0.700	0.781	0.861	
DIFFUSION FACTOR (ROTOR)	0.483	0.416	0.350	0.294	0.249	
DIFFUSION FACTOR (STATOR)	0.361	0.324	0.297	0.276	0.260	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.802	1.675	1.559	1.468	1.401	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.597	1.533	1.490	1.460	1.438	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00350	0.03427	0.03909	0.05285	0.08021	
SHOCK LOSS COEFFICIENT (STATOR)	0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.22E+06	.24E+06	.25E+06	.26E+06	.27E+06	
REYNLDS NUMBER (STATOR)	.35E+06	.36E+06	.37E+06	.37E+06	.38E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.01908	0.01974	0.02226	0.02526	0.02836
PROFILE LOSS COEFFICIENT (STATOR)	0.02895	0.02582	0.02339	0.02138	0.01968
SECONDARY LOSS COEFFICIENT (ROTOR)	0.00825				
SECONDARY LOSS COEFFICIENT (STATOR)	0.01241				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00942				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00621				
TOTAL LOSS COEFFICIENT (ROTOR)	0.08259				
TOTAL LOSS COEFFICIENT (STATOR)	0.04246				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.008

ROTOR ISENTROPIC EFFICIENCY 86.61 PERCENT

STAGE ISENTROPIC EFFICIENCY 83.62 PERCENT

STAGE PRESSURE RATIO 1.34

****

OVERALL PRESSURE RATIO 1.80

COMPRESSOR ISENTROPIC EFFICIENCY 83.11 PERCENT

COMPRESSOR POLYTROPIC EFFICIENCY 84.45 PERCENT

INPUT FILE FOR H.P. TURBINE

For Turbgeo Program

Turbofan-Turbojet Engine

```
HP TURBINE DESIGN
1                       !NUMBER OF STAGE
179.8                   !MASS FLOW
818706.0    !INLET TOTAL PRESSURE
1235                    !INLET TOTAL TEMPERATURE
0.01697     !FUEL AIR RATIO
220                     !INLET AXIAL VELOCITY
0.75                    !HUB/TIP RATIO
7900                    !RPM
0.9                     !POLYTROPIC EFFICIENCY
188                     !TEMPERATURE DROP
0.0                     !INLET AIR ANGLE
15.0                    !EXIT AIR ANGLE
1                       !DESIGN TYPE
1.8 2                   !ASPECT RATIO
0.81 0.7    !S/C RATIO
0.05                    !NOZZLE LOSS COEFFICIENT
```

OUTPUT FILE FOR THE H.P. TURBINE

From Turbgeo Program

Turbofan-Turbojet Engine

```
**********************************************************

     TURBINE PITCH-LINE DESIGN CALCULATION

     HP TURBINE DESIGN

          1 STAGE

   DESIGN INLET CONDITIONS

   MASS FLOW (Kg/s) ........................ 179.80
   INLET TOTAL PRESSURE (N/mý) ............ 818706.00
   OUTLET TOTAL PRESSURE (N/mý) ........... 373220.06
   INLET TOTAL TEMPERATURE (K) ............ 1235.00
   TEMPERATURE DROP (Tin-Tout) ............ 188.00
   AIR FUEL RATIO ......................... 0.01697
   AXIAL VELOCITY (m/s) ................... 220.00
   INLET HUB-TIP RATIO .................... 0.7500
   ROTATIONAL SPEED ....................... 7900.00
   ISENTROPIC EFFICIENCY (%) .............. 90.00
   INLET AIR ANGLE (degree) ............... 0.00
   OUTLET AIR ANGLE (degree) .............. 15.00
   NOZZLE ASPECT RATIO .................... 1.80
   ROTOR ASPECT RATIO ..................... 1.83
   NOZZLE/ROTOR SPACE-CHORD RATIO .........   0.81/0.70
   NOZZLE LOSS COEFFICIENT ................ 0.0500
   ROTOR LOSS COEFFICIENT ..............n.. 0.1314
   GAMMA .................................. 1.309
   SPECIFIC HEAT [Cpg] (J/kgK) ............ 1181.6208

        ANNULUS GEOMETRY

   TIP RADIUS (cm) .............. 52.09  55.65  56.54  61.35
   HUB RADIUS (cm) .............. 39.07  39.07  39.07  39.07
   NOZZLE AXIAL CHORD (cm) ...... 6.8853
   NOZZLE CHORD (cm) ............ 8.2244
   ROTOR AXIAL CHORD (cm) ....... 9.2872
   ROTOR CHORD (cm) ............. 10.8713
   INTER GAP (cm) ............... 1.7213
   OUTLET GAP (cm) .............. 2.3218
   OUTLET HUB-TIP RATIO ......... 0.6368
   ANNULUS INLET AREA (mý) ...... 0.3729
   NOZZLE EXIT  AREA (mý) ....... 0.2071
```

ROTOR EXIT AREA (mý) 0.6790
ANNULUS EXIT AREA (mý) 0.7029
ANNULUS FLARE ANGLE (degree) . 27.36
NUMBER OF BLADES (NOZZLE) 47
NUMBER OF BLADES (ROTOR) 41

ZWEIFEL S CORRELATION
NOZZLE/ROTOR SPACE-CHORD RATIO 1.52/ 1.32

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER 0.3257
NOZZLE OUTLET MACH NUMBER 0.8076
ROTOR INLET MACH NUMBER 0.3380
ROTOR INLET ROOT MACH NUMBER 0.5446
ROTOR EXIT MACH NUMBER 0.3654
MEAN BLADE SPEED (m/s) 415.37
FLOW COEFFICIENT 0.53
BLADE LOADING COEFFICIENT 1.29
REACTION %................................ 49.8131
REACTION AT THE ROOT %.................... 13.3111

NOZZLE ABS. INLET VELOCITY (m/s) 220.00
NOZZLE ABS. OUTLET VELOCITY (m/s)......... 524.26
ROTOR RELAT. INLET VELOCITY (m/s) 228.17
ROTOR RELAT. OUTLET VELOCITY (m/s) 522.85
ROTOR ABS. OUTLET VELOCITY (m/s).......... 227.76
ANNULUS EXIT AXIAL VELOCITY (m/s) 220.00

NOZZLE ABS. INLET AIR ANGLE (degree) 0.00
NOZZLE ABS. EXIT AIR ANGLE (degree) 65.19
ROTOR RELAT. INLET AIR ANGLE (degree) 15.38
ROTOR RELAT. EXIT AIR ANGLE (degree) 65.12
ROTOR ABS. EXIT AIR ANGLE (degree) 15.00

NOZZLE INLET BLADE ANGLE (degree) 0.00
NOZZLE EXIT BLADE ANGLE (degree) 52.58
ROTOR INLET BLADE ANGLE (degree) 15.38
ROTOR EXIT BLADE ANGLE (degree) 56.17

****
STAGE OVERALL CONDITION

PRESSURE RATIO 2.1936
PREDICTED ISENT. EFFICIENCY (%) 90.1549

****
This is the warning & error section which
gives information about flow conditions for
each appropriate section.
****

WARNING: ANNULUS INLET SUBSONIC FLOW

WARNING: ANNULUS OUTLET SUBSONIC FLOW

WARNING: NOZZLE OUTLET SUBSONIC FLOW

****
TURBINE OVERALL CONDITION

OVERALL PRESSURE RATIO (Pin/Pout) 2.1904
OVERALL ISENT. EFFICIENCY (%) 90.15
POLYTROPIC EFFICIENCY (%) 89.29
POWER OUTPUT (KJ/S) 39941.63

INPUT FILE FOR H.P. TURBINE

From Turbgeo to Turbdes Program

```
   1
HP TURBINE DESIGN                        STAGE          1
  39.94162
  179.8000
  7900.000
 0.9015488
 0.0000000E+00
  15.00000
  1.000000
 1.6969999E-02
  8.000000
  1.000000
```

```
0.0000000E+00
        0
        1
        1
        1
        1
        2
        1
        0
        0
        0
        0
        0
        0
 1235.000      0.0000000E+00  0.0000000E+00  0.0000000E+00  0.0000000E+00
 818.7060      0.0000000E+00  0.0000000E+00  0.0000000E+00  0.0000000E+00
0.0000000E+00  0.0000000E+00  0.0000000E+00  0.0000000E+00  0.0000000E+00
0.0000000E+00  0.0000000E+00   220.0000      0.0000000E+00  0.0000000E+00
0.0000000E+00  0.0000000E+00  0.0000000E+00  0.0000000E+00
 390.6720       390.6720       390.6720       390.6720
 520.8959       556.5266       565.4343       613.4944
       47            41
0.0000000E+00  0.0000000E+00
0.0000000E+00  0.0000000E+00
0.0000000E+00  0.0000000E+00
0.2500000      0.2500000
 1.800000       1.828595
 68.85281       92.87161
```

OUTPUT FILE FOR H.P. TURBINE

From Turbdes Program

```
                    Free Vortex Turbine Design Program
                    ----------------------------------

          HP TURBINE DESIGN                   STAGE          1
0         Input Data

      Inlet Mass Flow Rate                  179.80 kg/s
      Power Output                         39.9416 MW
      Shaft Rotational Speed                7900.0 rev/min
      Inlet  Swirl Angle                      0.00 deg
      Turbine T/T Isentropic Efficiency       0.90
      Outlet Swirl Angle                     15.00 deg
      Inlet Fuel/Air Ratio                  0.0170
      Work Done Factor                        1.00

             P(inf)    T(inf)
             (kPa)      (K)
     Tip     818.7     1235.0
     Mean    818.7     1235.0
     Root    818.7     1235.0

         Blade Stress Input Information

     Metal Density  8.00      Mg/cu.m   Taper Factor 1.000   Shroud Equivalent Length  0.00 mm
NO=          1
N=          2
          Plane Number          1             2             3              4
          Axial Distance        0.0          68.9         103.3          196.2    (mm)
          Tip  Radius         520.90        556.53        565.43         613.49    (mm)
          Mean Radius         455.78        473.60        478.05         502.08    (mm)
          Root Radius         390.67        390.67        390.67         390.67    (mm)

          N.G.V. Aerofoil Data                       Rotor Aerofoil Data
          True  Chord       81.87  (mm)              True  Chord       *****  (mm)
          Axial Chord       68.85  (mm)              Axial Chord       92.87  (mm)
          Max Thickness     16.36  (mm)              Max Thickness     20.09  (mm)
          T.E. Thickness     1.27  (mm)              T.E. Thickness     1.45  (mm)
                                                     Number of Seals    0
                                                 Tip Seal Clearance     0.25  (mm)

          Ainley,Mathieson,Dunham & Came Loss Coefficients
          N.G.V.                                        Rotor
          Profile Loss    (YP)   0.031                  Profile Loss    (YP)   0.041
          Secondary Loss  (YS)   0.038                  Secondary Loss  (YS)   0.046
          Tip             (YK)   0.000                  Tip             (YK)   0.014
          Total + R.E.           0.068                  Total + R.E.           0.101
          Reynolds Number        1.54E+06               Reynolds Number        1.41E+06

     Number of N.G.V.'s  47   Space/Axial Chord Ratio 0.92   Axial Chord Aspect Ratio 2.41
                              Space/Chord Ratio       0.77   Aspect Ratio             2.03

     Number of Rotor 's  41   Space/Axial Chord Ratio 0.81   Axial Chord Aspect Ratio 2.14
                              Space/Chord Ratio       0.75   Aspect Ratio             1.98

          Nozzle Guide Vane Outlet Triangles,
```

	V(0) (m/s)	V(1) (m/s)	VW(0) (m/s)	ALFA(0) (deg)	ALFA(1) (deg)	MA(0) (-)	MA(1) (-)	VA(0,1) (m/s)
Tip	473.6	234.2	417.6	61.84	-17.41	0.722	0.357	223.5
Mean	539.2	239.2	490.7	65.51	20.88	0.832	0.369	223.5
Root	635.5	351.8	594.9	69.41	50.55	1.001	0.554	223.5

Rotor Outlet Triangles

	V(3) (m/s)	V(2) (m/s)	VW(3) (m/s)	ALFA(3) (deg)	ALFA(2) (deg)	MA(3) (-)	MA(2) (-)	VA(2,3) (m/s)
Tip	213.4	572.7	45.7	12.37	68.65	0.343	0.919	208.5
Mean	215.8	506.2	55.9	15.00	65.68	0.346	0.811	208.5
Root	220.5	446.6	71.8	19.00	62.18	0.353	0.714	208.5

Stagnation Temperatures & Pressures

	T(0) (K)	T(1) (K)	T(2) (K)	T(3) (K)	P(0) (kPa)	P(1) (kPa)	P(2) (kPa)	P(3) (kPa)
Tip	1235.0	1164.5	1164.5	1046.9	803.3	628.1	603.5	386.7
Mean	1235.0	1137.8	1137.8	1050.5	799.5	567.5	548.8	393.1
Root	1235.0	1118.4	1118.4	1055.6	793.8	524.4	510.2	400.7

	Velocity Ratios (0)/(inf) (-)	(2)/(1) (-)	Deflections (inf,0) (deg)	(1,2) (deg)	Blade Stress Tensile (MPa)	Bending (MPa)	Outlet Static Pressure (kPa)
Tip	2.15	2.44	61.8	51.2	0.0	0.0	358.2
Mean	2.45	2.12	65.5	86.6	0.0	16.5	363.6
Root	2.89	1.27	69.4	112.7	0.0	43.9	369.5

Stage Parameters

	u (m/s)	dVw (m/s)	dH/u.sq (-)	Reaction (%)	Zweifel Lift Coef Stator (-)	Rotor (-)
Tip	487.7	463.3	0.95	60.4	0.70	0.47
Mean	405.4	546.6	1.35	44.9	0.57	0.61
Root	323.2	666.6	2.06	17.6	0.45	0.78

ETA(TT) = 92.4 %   CP dT/T = 346.7   N/RT = 0.0
ETA(TS) = 84.3 %   Va/u = 0.53

Warning:Mach Number(s) in excess of Unity

****

OVERALL PRESSURE RATIO.................. 2.083
TURBINE ISENTROPIC EFFICIENCY.......... 92.835
TURBINE POLYTROPIC EFFICIENCY.......... 92.222

INPUT FILE FOR THE L.P. TURBINE

For Turbgeo Program

Turbofan-Turbojet Engine
Low pressure Turbine

```
LP TURBINE DESIGN
2                    !NUMBER OF STAGE
191                  !MASS FLOW
475700.6     !INLET TOTAL PRESSURE
1021                 !INLET TOTAL TEMPERATURE
0.01595              !FUEL AIR RATIO
175                  !INLET AXIAL VELOCITY
0.7                  !HUB/TIP RATIO
3800                 !RPM
0.9                  !POLYTROPIC EFFICIENCY
92 92                !TEMPERATURE DROP
0 35                 !INLET AIR ANGLE
33 5                 !EXIT AIR ANGLE
2                    !DESIGN TYPE
2.0 2.2 2.3 2.4      !ASPECT RATIO
0.8 0.8 0.8 0.8 !S/C RATIO
0.05                 !NOZZLE LOSS COEFFICIENT
```

OUTPUT FILE FOR THE L.P. TURBINE

From Turbgeo Program

****

TURBINE PITCH-LINE DESIGN CALCULATION

LP TURBINE DESIGN

1 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s)	191.00
INLET TOTAL PRESSURE (N/mý)	475700.59
OUTLET TOTAL PRESSURE (N/mý)	308950.13
INLET TOTAL TEMPERATURE (K)	1021.00
TEMPERATURE DROP (Tin-Tout)	92.00
AIR FUEL RATIO	0.01595
AXIAL VELOCITY (m/s)	175.00
INLET HUB-TIP RATIO	0.7000
ROTATIONAL SPEED	3800.00
ISENTROPIC EFFICIENCY (%)	90.00
INLET AIR ANGLE (degree)	0.00
OUTLET AIR ANGLE (degree)	35.00
NOZZLE ASPECT RATIO	2.00
ROTOR ASPECT RATIO	2.78
NOZZLE/ROTOR SPACE-CHORD RATIO	0.80/0.80
NOZZLE LOSS COEFFICIENT	0.0500
ROTOR LOSS COEFFICIENT	0.1048
GAMMA	1.323
SPECIFIC HEAT [Cpg] (J/kgK)	1153.7493

ANNULUS GEOMETRY

TIP RADIUS (cm)	66.08	68.04	68.54	70.41
HUB RADIUS (cm)	46.25	44.29	43.80	41.92
NOZZLE AXIAL CHORD (cm)	9.2370			
NOZZLE CHORD (cm)	10.8948			
ROTOR AXIAL CHORD (cm)	8.7923			
ROTOR CHORD (cm)	9.5794			
INTER GAP (cm)	2.3093			
OUTLET GAP (cm)	2.1981			
OUTLET HUB-TIP RATIO	0.5955			
ANNULUS INLET AREA (mý)	0.6996			
NOZZLE EXIT AREA (mý)	0.3729			
ROTOR EXIT AREA (mý)	0.8234			
ANNULUS EXIT AREA (mý)	1.0051			
ANNULUS FLARE ANGLE (degree) .	24.04			
NUMBER OF BLADES (NOZZLE)	40			
NUMBER OF BLADES (ROTOR)	46			

ZWEIFEL S CORRELATION

NOZZLE/ROTOR SPACE-CHORD RATIO 1.48/ 0.99

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER	0.2828
NOZZLE OUTLET MACH NUMBER	0.6532
ROTOR INLET MACH NUMBER	0.3525
ROTOR INLET ROOT MACH NUMBER	0.5392
ROTOR EXIT MACH NUMBER	0.3625
MEAN BLADE SPEED (m/s)	223.50
FLOW COEFFICIENT	0.78
BLADE LOADING COEFFICIENT	2.12
REACTION %................................	48.5823
REACTION AT THE ROOT %....................	12.0369
NOZZLE ABS. INLET VELOCITY (m/s)	175.00
NOZZLE ABS. OUTLET VELOCITY (m/s).........	393.44
ROTOR RELAT. INLET VELOCITY (m/s)	217.33
ROTOR RELAT. OUTLET VELOCITY (m/s)	387.77
ROTOR ABS. OUTLET VELOCITY (m/s)..........	213.64
ANNULUS EXIT AXIAL VELOCITY (m/s)	175.00
NOZZLE ABS. INLET AIR ANGLE (degree)	0.00
NOZZLE ABS. EXIT AIR ANGLE (degree)	63.59
ROTOR RELAT. INLET AIR ANGLE (degree)	36.37

ROTOR RELAT. EXIT AIR ANGLE (degree) 63.17
ROTOR ABS. EXIT AIR ANGLE (degree) 35.00

NOZZLE INLET BLADE ANGLE (degree) 0.00
NOZZLE EXIT BLADE ANGLE (degree) 51.36
ROTOR INLET BLADE ANGLE (degree) 36.37
ROTOR EXIT BLADE ANGLE (degree) 58.01

****

STAGE OVERALL CONDITION

PRESSURE RATIO 1.5397
PREDICTED ISENT. EFFICIENCY (%) 90.1708

****
This is the warning & error section which
gives information about flow conditions for
each appropriate section.
****

WARNING: ANNULUS INLET SUBSONIC FLOW

WARNING: ANNULUS OUTLET SUBSONIC FLOW

WARNING: NOZZLE OUTLET SUBSONIC FLOW

****

TURBINE PITCH-LINE DESIGN CALCULATION

LP TURBINE DESIGN

2 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s) 191.00
INLET TOTAL PRESSURE (N/mý) 308950.13
OUTLET TOTAL PRESSURE (N/mý) 193359.92
INLET TOTAL TEMPERATURE (K) 929.00
TEMPERATURE DROP (Tin-Tout) 92.00
AIR FUEL RATIO 0.01595
AXIAL VELOCITY (m/s) 175.00
INLET HUB-TIP RATIO 0.5955
ROTATIONAL SPEED 3800.00
ISENTROPIC EFFICIENCY (%) 90.00
INLET AIR ANGLE (degree) 35.00
OUTLET AIR ANGLE (degree) 33.00
NOZZLE ASPECT RATIO 2.20
ROTOR ASPECT RATIO 2.62
NOZZLE/ROTOR SPACE-CHORD RATIO 0.80/0.80
NOZZLE LOSS COEFFICIENT 0.0500
ROTOR LOSS COEFFICIENT 0.1062
GAMMA 1.331
SPECIFIC HEAT [Cpg] (J/kgK) 1131.9668

ANNULUS GEOMETRY

TIP RADIUS (cm) 70.41 73.17 73.86 76.74
HUB RADIUS (cm) 41.92 39.16 38.47 35.59
NOZZLE AXIAL CHORD (cm) 12.9376
NOZZLE CHORD (cm) 14.2021
ROTOR AXIAL CHORD (cm) 13.4802
ROTOR CHORD (cm) 14.6081
INTER GAP (cm) 3.2344
OUTLET GAP (cm) 3.3701
OUTLET HUB-TIP RATIO 0.4638
ANNULUS INLET AREA (mý) 1.0051
NOZZLE EXIT AREA (mý) 0.5339
ROTOR EXIT AREA (mý) 1.2177
ANNULUS EXIT AREA (mý) 1.4520
ANNULUS FLARE ANGLE (degree) . 24.10
NUMBER OF BLADES (NOZZLE) 31
NUMBER OF BLADES (ROTOR) 30

ZWEIFEL S CORRELATION
NOZZLE/ROTOR SPACE-CHORD RATIO 1.02/ 0.97

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER 0.3625
NOZZLE OUTLET MACH NUMBER 0.6855
ROTOR INLET MACH NUMBER 0.3688
ROTOR INLET ROOT MACH NUMBER 0.7006
ROTOR EXIT MACH NUMBER 0.3721
MEAN BLADE SPEED (m/s) 223.50
FLOW COEFFICIENT 0.78
BLADE LOADING COEFFICIENT 2.08
REACTION %................................. 46.6106
REACTION AT THE ROOT %.................... -9.4473

NOZZLE ABS. INLET VELOCITY (m/s) 213.64
NOZZLE ABS. OUTLET VELOCITY (m/s)......... 393.37
ROTOR RELAT. INLET VELOCITY (m/s) 217.29
ROTOR RELAT. OUTLET VELOCITY (m/s) 379.86
ROTOR ABS. OUTLET VELOCITY (m/s).......... 208.66
ANNULUS EXIT AXIAL VELOCITY (m/s) 175.00

NOZZLE ABS. INLET AIR ANGLE (degree) 35.00
NOZZLE ABS. EXIT AIR ANGLE (degree) 63.58
ROTOR RELAT. INLET AIR ANGLE (degree) 36.35
ROTOR RELAT. EXIT AIR ANGLE (degree) 62.57
ROTOR ABS. EXIT AIR ANGLE (degree) 33.00

NOZZLE INLET BLADE ANGLE (degree) 35.00
NOZZLE EXIT BLADE ANGLE (degree) 58.08
ROTOR INLET BLADE ANGLE (degree) 36.35
ROTOR EXIT BLADE ANGLE (degree) 57.52

****

STAGE OVERALL CONDITION

PRESSURE RATIO 1.5978
PREDICTED ISENT. EFFICIENCY (%) 90.1835

****

This is the warning & error section which
gives information about flow conditions for
each appropriate section.

****

WARNING: ANNULUS INLET SUBSONIC FLOW

WARNING: ANNULUS OUTLET SUBSONIC FLOW

WARNING: NOZZLE OUTLET SUBSONIC FLOW

****

TURBINE OVERALL CONDITION

OVERALL PRESSURE RATIO (Pin/Pout) 2.4556
OVERALL ISENT. EFFICIENCY (%) 90.69
POLYTROPIC EFFICIENCY (%) 89.70
POWER OUTPUT (KJ/S) 39781.84

INPUT FILE FOR L.P. TURBINE

From Turbgeo to Turbdes Program

```
    2
LP TURBINE DESIGN          STAGE     1
 20.27368
 191.0000
 3800.000
0.9017078
0.0000000E+00
 35.00000
 1.000000
1.5950000E-02
 8.000000
 1.000000
0.0000000E+00
    0
    1
    1
    2
```

```
      2
      2
      1
      1
      0
      0
      0
      0
      0
 1021.000    0.0000000E+00 0.0000000E+00 0.0000000E+00 0.0000000E+00
 475.7006    0.0000000E+00 0.0000000E+00 0.0000000E+00 0.0000000E+00
0.0000000E+00 0.0000000E+00 0.0000000E+00 0.0000000E+00 0.0000000E+00
0.0000000E+00 0.0000000E+00  175.0000   0.0000000E+00 0.0000000E+00
0.0000000E+00 0.0000000E+00 0.0000000E+00 0.0000000E+00
 462.5425    442.8790    437.9631    419.2462
 660.7751    680.4387    685.3546    704.0714
     40       46
0.0000000E+00 0.0000000E+00
0.0000000E+00 0.0000000E+00
0.0000000E+00 0.0000000E+00
0.2500000    0.2500000
 2.000000    2.777917
 92.37039    87.92295
      0
LP TURBINE DESIGN              STAGE       2
 19.89092
 191.0000
 3800.000
0.9018351
0.0000000E+00
 33.00000
 1.000000
1.5950000E-02
 8.000000
 1.000000
0.0000000E+00
      0
      1
      1
      2
      2
      2
      1
      2
      1
      0
      0
      0
      0
0.0000000E+00 0.0000000E+00 0.0000000E+00 0.0000000E+00
 419.2462    391.6245    384.7191    355.9389
 704.0714    731.6931    738.5985    767.3787
     31       30
0.0000000E+00 0.0000000E+00
0.0000000E+00 0.0000000E+00
0.0000000E+00 0.0000000E+00
0.2500000    0.2500000
 2.200000    2.619496
 129.3765    134.8023
      0
```

OUTPUT FILE FOR L.P. TURBINE

From Turbdes Program

```
                      Free Vortex Turbine Design Program
                      ----------------------------------

          LP TURBINE DESIGN                  STAGE           1
 0        Input Data

      Inlet Mass Flow Rate                 191.00 kg/s
      Power Output                        20.2737 MW
      Shaft Rotational Speed               3800.0 rev/min
      Inlet  Swirl Angle                     0.00 deg
      Turbine T/T Isentropic Efficiency      0.90
      Outlet Swirl Angle                    35.00 deg
      Inlet Fuel/Air Ratio                 0.0159
      Work Done Factor                       1.00

            P(inf)    T(inf)
            (kPa)      (K)
    Tip     475.7     1021.0
    Mean    475.7     1021.0
    Root    475.7     1021.0
```

Blade Stress Input Information

Metal Density 8.00 Mg/cu.m Taper Factor 1.000 Shroud Equivalent Length 0.00 mm

NO= 1

N= 2

Plane Number	1	2	3	4	
Axial Distance	0.0	92.4	138.6	226.5	(mm)
Tip Radius	660.78	680.44	685.35	704.07	(mm)
Mean Radius	561.66	561.66	561.66	561.66	(mm)
Root Radius	462.54	442.88	437.96	419.25	(mm)

N.G.V. Aerofoil Data			Rotor Aerofoil Data		
True Chord	*****	(mm)	True Chord	90.35	(mm)
Axial Chord	92.37	(mm)	Axial Chord	87.92	(mm)
Max Thickness	21.74	(mm)	Max Thickness	18.07	(mm)
T.E. Thickness	1.76	(mm)	T.E. Thickness	1.53	(mm)
			Number of Seals	0	
			Tip Seal Clearance	0.25	(mm)

Ainley,Mathieson,Dunham & Came Loss Coefficients

N.G.V.			Rotor		
Profile Loss	(YP)	0.029	Profile Loss	(YP)	0.061
Secondary Loss	(YS)	0.034	Secondary Loss	(YS)	0.045
Tip	(YK)	0.000	Tip	(YK)	0.011
Total + R.E.		0.062	Total + R.E.		0.118
Reynolds Number		1.35E+06	Reynolds Number		9.44E+05

Number of N.G.V.'s 40 Space/Axial Chord Ratio 0.96 Axial Chord Aspect Ratio 2.57
Space/Chord Ratio 0.81 Aspect Ratio 2.18

Number of Rotor 's 46 Space/Axial Chord Ratio 0.87 Axial Chord Aspect Ratio 3.03
Space/Chord Ratio 0.85 Aspect Ratio 2.95

Nozzle Guide Vane Outlet Triangles,

	V(0) (m/s)	V(1) (m/s)	VW(0) (m/s)	ALFA(0) (deg)	ALFA(1) (deg)	MA(0) (-)	MA(1) (-)	VA(0,1) (m/s)
Tip	339.8	175.0	291.7	59.14	5.01	0.558	0.288	174.3
Mean	394.1	217.4	353.4	63.75	36.70	0.653	0.360	174.3
Root	480.9	327.8	448.2	68.75	57.88	0.811	0.553	174.3

Rotor Outlet Triangles

	V(3) (m/s)	V(2) (m/s)	VW(3) (m/s)	ALFA(3) (deg)	ALFA(2) (deg)	MA(3) (-)	MA(2) (-)	VA(2,3) (m/s)
Tip	198.7	411.7	96.9	29.19	65.08	0.337	0.698	173.5
Mean	211.8	386.1	121.5	35.00	63.30	0.359	0.655	173.5
Root	237.9	375.7	162.7	43.17	62.50	0.404	0.639	173.5

Stagnation Temperatures & Pressures

	T(0) (K)	T(1) (K)	T(2) (K)	T(3) (K)	P(0) (kPa)	P(1) (kPa)	P(2) (kPa)	P(3) (kPa)
Tip	1021.0	984.6	984.6	928.7	470.3	405.9	393.5	310.6
Mean	1021.0	974.6	974.6	929.9	468.7	388.2	377.4	311.9
Root	1021.0	967.8	967.8	931.5	465.8	375.0	365.1	312.6

	Velocity Ratios		Deflections		Blade Stress		Outlet Static
	(0)/(inf) (-)	(2)/(1) (-)	(inf,0) (deg)	(1,2) (deg)	Tensile (MPa)	Bending (MPa)	Pressure (kPa)
Tip	1.94	2.35	59.1	70.1	0.0	0.0	288.2
Mean	2.25	1.78	63.8	100.0	0.0	20.1	286.6
Root	2.75	1.15	68.8	120.4	0.0	52.8	280.9

Stage Parameters

	u (m/s)	dVw (m/s)	dH/u.sq (-)	Reaction (%)	Zweifel Lift Coef Stator (-)	Rotor (-)
Tip	276.5	388.6	1.41	64.6	0.88	0.76
Mean	223.5	474.9	2.12	48.0	0.68	0.87
Root	170.6	611.0	3.58	16.2	0.49	0.92

ETA(TT) = 89.9 % CP dT/T = 194.9 N/RT = 0.0
ETA(TS) = 75.6 % Va/u = 0.78

Free Vortex Turbine Design Program

LP TURBINE DESIGN STAGE 2
0 Input Data

Inlet Mass Flow Rate	191.00 kg/s
Power Output	19.8909 MW
Shaft Rotational Speed	3800.0 rev/min
Inlet Swirl Angle	35.00 deg
Turbine T/T Isentropic Efficiency	0.90

```
Outlet Swirl Angle                        33.00 deg
Inlet Fuel/Air Ratio                     0.0159
Work Done Factor                          1.00

       P(inf)   T(inf)
       (kPa)     (K)
Tip    311.9    931.5
Mean   311.9    929.9
Root   311.9    928.7

     Blade Stress Input Information

   Metal Density  8.00       Mg/cu.m   Taper Factor 1.000   Shroud Equivalent Length  0.00 mm
NO=         1
N=         2
       Plane Number        1            2            3            4
       Axial Distance     288.6        418.0        482.6        617.4    (mm)
       Tip  Radius        704.07       731.69       738.60       767.38   (mm)
       Mean Radius        561.66       561.66       561.66       561.66   (mm)
       Root Radius        419.25       391.62       384.72       355.94   (mm)

       N.G.V. Aerofoil Data                   Rotor Aerofoil Data
       True  Chord       *****  (mm)          True  Chord       *****  (mm)
       Axial Chord       *****  (mm)          Axial Chord       *****  (mm)
       Max Thickness     30.50  (mm)          Max Thickness     27.66  (mm)
       T.E. Thickness     2.28  (mm)          T.E. Thickness     2.35  (mm)
                                              Number of Seals    0
                                            Tip Seal Clearance   0.25  (mm)

       Ainley,Mathieson,Dunham & Came Loss Coefficients
       N.G.V.                                    Rotor
       Profile Loss    (YP)   0.056              Profile Loss    (YP)   0.060
       Secondary Loss  (YS)   0.061              Secondary Loss  (YS)   0.049
       Tip             (YK)   0.000              Tip             (YK)   0.009
       Total + R.E.           0.118              Total + R.E.           0.117
       Reynolds Number        1.39E+06           Reynolds Number        1.04E+06

   Number of N.G.V.'s  31    Space/Axial Chord Ratio 0.88    Axial Chord Aspect Ratio 2.63
                             Space/Chord Ratio       0.75    Aspect Ratio             2.23

   Number of Rotor 's  30    Space/Axial Chord Ratio 0.87    Axial Chord Aspect Ratio 2.84
                             Space/Chord Ratio       0.85    Aspect Ratio             2.77
```

Nozzle Guide Vane Outlet Triangles,

	V(0) (m/s)	V(1) (m/s)	VW(0) (m/s)	ALFA(0) (deg)	ALFA(1) (deg)	MA(0) (-)	MA(1) (-)	VA(0,1) (m/s)
Tip	322.1	176.6	270.9	57.25	-9.36	0.553	0.303	174.3
Mean	393.6	217.1	352.9	63.72	36.60	0.685	0.378	174.3
Root	535.3	398.9	506.1	71.00	64.09	0.963	0.717	174.3

Rotor Outlet Triangles

	V(3) (m/s)	V(2) (m/s)	VW(3) (m/s)	ALFA(3) (deg)	ALFA(2) (deg)	MA(3) (-)	MA(2) (-)	VA(2,3) (m/s)
Tip	192.7	420.1	82.7	25.42	65.52	0.344	0.749	174.1
Mean	207.5	378.9	113.0	33.00	62.65	0.370	0.676	174.1
Root	249.2	369.3	178.4	45.70	61.88	0.445	0.660	174.1

Stagnation Temperatures & Pressures

	T(0) (K)	T(1) (K)	T(2) (K)	T(3) (K)	P(0) (kPa)	P(1) (kPa)	P(2) (kPa)	P(3) (kPa)
Tip	928.7	897.0	897.0	836.1	305.4	266.0	256.9	194.1
Mean	929.9	882.7	882.7	838.8	302.7	246.0	238.9	195.0
Root	931.5	875.8	875.8	843.3	296.7	232.1	225.6	194.1

	Velocity Ratios (0)/(inf) (-)	Velocity Ratios (2)/(1) (-)	Deflections (inf,0) (deg)	Deflections (1,2) (deg)	Blade Stress Tensile (MPa)	Blade Stress Bending (MPa)	Outlet Static Pressure (kPa)
Tip	1.52	2.38	86.4	56.2	0.0	0.0	179.5
Mean	1.86	1.75	98.7	99.2	0.0	12.4	178.1
Root	2.53	0.93	114.2	126.0	0.0	32.1	170.3

Stage Parameters

	u (m/s)	dVw (m/s)	dH/u.sq (-)	Reaction (%)	Zweifel Lift Coef Stator (-)	Zweifel Lift Coef Rotor (-)
Tip	299.6	353.6	1.18	68.6	1.20	0.70
Mean	223.5	465.9	2.08	46.3	0.84	0.88
Root	147.4	684.5	4.64	****	0.50	0.94

```
ETA(TT) = 87.5 %        CP dT/T = 205.7        N/RT =   0.0
ETA(TS) = 74.2 %           Va/u =    0.78


     ************************************************************

     OVERALL PRESSURE RATIO.................  2.440
     TURBINE ISENTROPIC EFFICIENCY..........  88.925
     TURBINE POLYTROPIC EFFICIENCY..........  87.756
```

Effect of Variable Geometry

INPUT FILE FOR THE L.P. TURBINE

For Turbgeo Program

Effect of Variable Geometry
Turbofan-Turbojet Engine
Low pressure Turbine
Design point is The Subsonic Point (Altitude 9150 m, Mach Number 0.95).

```
LP TURBINE DESIGN
2               !NUMBER OF STAGE
103             !MASS FLOW
247496    !INLET TOTAL PRESSURE
957             !INLET TOTAL TEMPERATURE
0.01479   !FUEL AIR RATIO
170             !INLET AXIAL VELOCITY
0.7             !HUB/TIP RATIO
3800            !RPM
0.9             !POLYTROPIC EFFICIENCY
92.5 92.5 !TEMPERATURE DROP
0 35            !INLET AIR ANGLE
33 5            !EXIT AIR ANGLE
2               !DESIGN TYPE
2.0 2.2 2.3 2.4 !ASPECT RATIO
0.8 0.8 0.8 0.8 !S/C RATIO
0.05            !NOZZLE LOSS COEFFICIENT
```

Effect of Variable Geometry

OUTPUT FILE FOR THE L.P. TURBINE

From Turbgeo Program

Effect of Variable Geometry
Turbofan-Turbojet Engine
Low pressure Turbine
Design point is The Subsonic Point (Altitude 9150 m, Mach Number 0.95).

****

TURBINE PITCH-LINE DESIGN CALCULATION

LP TURBINE DESIGN

1 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s) 102.70
INLET TOTAL PRESSURE (N/mý) 247496.00
OUTLET TOTAL PRESSURE (N/mý) 156552.33
INLET TOTAL TEMPERATURE (K) 958.00
TEMPERATURE DROP (Tin-Tout) 92.50
AIR FUEL RATIO 0.01479
AXIAL VELOCITY (m/s) 170.00
INLET HUB-TIP RATIO 0.7000
ROTATIONAL SPEED 3800.00
ISENTROPIC EFFICIENCY (%) 90.00
INLET AIR ANGLE (degree) 0.00
OUTLET AIR ANGLE (degree) 35.00
NOZZLE ASPECT RATIO 2.00
ROTOR ASPECT RATIO 2.82
NOZZLE/ROTOR SPACE-CHORD RATIO 0.80/0.80
NOZZLE LOSS COEFFICIENT 0.0500
ROTOR LOSS COEFFICIENT 0.1072
GAMMA 1.329

SPECIFIC HEAT [Cpg] (J/kgK) 1136.8281

ANNULUS GEOMETRY

TIP RADIUS (cm) 66.02 68.12 68.64 70.63
HUB RADIUS (cm) 46.21 44.12 43.60 41.60
NOZZLE AXIAL CHORD (cm) 9.2367
NOZZLE CHORD (cm) 10.9505
ROTOR AXIAL CHORD (cm) 8.7819
ROTOR CHORD (cm) 9.5863
INTER GAP (cm) 2.3092
OUTLET GAP (cm) 2.1955
OUTLET HUB-TIP RATIO 0.5890
ANNULUS INLET AREA (mý) 0.6984
NOZZLE EXIT AREA (mý) 0.3681
ROTOR EXIT AREA (mý) 0.8384
ANNULUS EXIT AREA (mý) 1.0235
ANNULUS FLARE ANGLE (degree) . 25.55
NUMBER OF BLADES (NOZZLE) 40
NUMBER OF BLADES (ROTOR) 45

ZWEIFEL S CORRELATION
NOZZLE/ROTOR SPACE-CHORD RATIO 1.49/ 0.99

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER 0.2830
NOZZLE OUTLET MACH NUMBER 0.6698
ROTOR INLET MACH NUMBER 0.3562
ROTOR INLET ROOT MACH NUMBER 0.5545
ROTOR EXIT MACH NUMBER 0.3640
MEAN BLADE SPEED (m/s) 223.31
FLOW COEFFICIENT 0.76
BLADE LOADING COEFFICIENT 2.11
REACTION %............................... 47.8702
REACTION AT THE ROOT %.................. 10.5848

NOZZLE ABS. INLET VELOCITY (m/s) 170.00
NOZZLE ABS. OUTLET VELOCITY (m/s)......... 390.78
ROTOR RELAT. INLET VELOCITY (m/s) 213.13
ROTOR RELAT. OUTLET VELOCITY (m/s) 382.23
ROTOR ABS. OUTLET VELOCITY (m/s).......... 207.53
ANNULUS EXIT AXIAL VELOCITY (m/s) 170.00

NOZZLE ABS. INLET AIR ANGLE (degree) 0.00
NOZZLE ABS. EXIT AIR ANGLE (degree) 64.21
ROTOR RELAT. INLET AIR ANGLE (degree) 37.10
ROTOR RELAT. EXIT AIR ANGLE (degree) 63.59
ROTOR ABS. EXIT AIR ANGLE (degree) 35.00

NOZZLE INLET BLADE ANGLE (degree) 0.00
NOZZLE EXIT BLADE ANGLE (degree) 51.86
ROTOR INLET BLADE ANGLE (degree) 37.10
ROTOR EXIT BLADE ANGLE (degree) 58.49

****
STAGE OVERALL CONDITION

PRESSURE RATIO 1.5809
PREDICTED ISENT. EFFICIENCY (%) 90.1747

****
This is the warning & error section which
gives information about flow conditions for
each appropriate section.
****
WARNING: ANNULUS INLET SUBSONIC FLOW

WARNING: ANNULUS OUTLET SUBSONIC FLOW

WARNING: NOZZLE OUTLET SUBSONIC FLOW
****

TURBINE PITCH-LINE DESIGN CALCULATION

LP TURBINE DESIGN

2 STAGE

DESIGN INLET CONDITIONS

MASS FLOW (Kg/s) 102.70
INLET TOTAL PRESSURE (N/mý) 156552.33
OUTLET TOTAL PRESSURE (N/mý) 94884.18
INLET TOTAL TEMPERATURE (K) 865.50
TEMPERATURE DROP (Tin-Tout) 92.50
AIR FUEL RATIO 0.01479
AXIAL VELOCITY (m/s) 170.00
INLET HUB-TIP RATIO 0.5890
ROTATIONAL SPEED 3800.00
ISENTROPIC EFFICIENCY (%) 90.00
INLET AIR ANGLE (degree) 35.00

OUTLET AIR ANGLE (degree) 33.00
NOZZLE ASPECT RATIO 2.20
ROTOR ASPECT RATIO 2.66
NOZZLE/ROTOR SPACE-CHORD RATIO 0.80/0.80
NOZZLE LOSS COEFFICIENT 0.0500
ROTOR LOSS COEFFICIENT 0.1085
GAMMA 1.338
SPECIFIC HEAT [Cpg] (J/kgK) 1114.4475

ANNULUS GEOMETRY

TIP RADIUS (cm) 70.63 73.67 74.43 77.59
HUB RADIUS (cm) 41.60 38.57 37.81 34.64
NOZZLE AXIAL CHORD (cm) 13.2005
NOZZLE CHORD (cm) 14.5736
ROTOR AXIAL CHORD (cm) 13.7652
ROTOR CHORD (cm) 14.9499
INTER GAP (cm) 3.3001
OUTLET GAP (cm) 3.4413
OUTLET HUB-TIP RATIO 0.4465
ANNULUS INLET AREA (mý) 1.0235
NOZZLE EXIT AREA (mý) 0.5392
ROTOR EXIT AREA (mý) 1.2700
ANNULUS EXIT AREA (mý) 1.5143
ANNULUS FLARE ANGLE (degree) . 25.90
NUMBER OF BLADES (NOZZLE) 30
NUMBER OF BLADES (ROTOR) 29

ZWEIFEL S CORRELATION
NOZZLE/ROTOR SPACE-CHORD RATIO 1.04/ 0.98

PITCH LINE GAS PARAMETERS

NOZZLE INLET MACH NUMBER 0.3640
NOZZLE OUTLET MACH NUMBER 0.7047
ROTOR INLET MACH NUMBER 0.3734
ROTOR INLET ROOT MACH NUMBER 0.7397
ROTOR EXIT MACH NUMBER 0.3753
MEAN BLADE SPEED (m/s) 223.31
FLOW COEFFICIENT 0.76
BLADE LOADING COEFFICIENT 2.07
REACTION %................................ 46.0786
REACTION AT THE ROOT %.................... -12.6543

NOZZLE ABS. INLET VELOCITY (m/s) 207.53
NOZZLE ABS. OUTLET VELOCITY (m/s)......... 390.20
ROTOR RELAT. INLET VELOCITY (m/s) 212.75
ROTOR RELAT. OUTLET VELOCITY (m/s) 374.52
ROTOR ABS. OUTLET VELOCITY (m/s).......... 202.70
ANNULUS EXIT AXIAL VELOCITY (m/s) 170.00

NOZZLE ABS. INLET AIR ANGLE (degree) 35.00
NOZZLE ABS. EXIT AIR ANGLE (degree) 64.17
ROTOR RELAT. INLET AIR ANGLE (degree) 36.96
ROTOR RELAT. EXIT AIR ANGLE (degree) 63.00
ROTOR ABS. EXIT AIR ANGLE (degree) 33.00

NOZZLE INLET BLADE ANGLE (degree) 35.00
NOZZLE EXIT BLADE ANGLE (degree) 58.56
ROTOR INLET BLADE ANGLE (degree) 36.96
ROTOR EXIT BLADE ANGLE (degree) 57.99

****

STAGE OVERALL CONDITION

PRESSURE RATIO 1.6499
PREDICTED ISENT. EFFICIENCY (%) 90.1894

****

This is the warning & error section which
gives information about flow conditions for
each appropriate section.

****

WARNING: ANNULUS INLET SUBSONIC FLOW

WARNING: ANNULUS OUTLET SUBSONIC FLOW

WARNING: NOZZLE OUTLET SUBSONIC FLow

****

TURBINE OVERALL CONDITION

OVERALL PRESSURE RATIO (Pin/Pout) 2.6030
OVERALL ISENT. EFFICIENCY (%) 90.74
POLYTROPIC EFFICIENCY (%) 89.67
POWER OUTPUT (KJ/S) 21173.95

Effect of Variable Geometry

INPUT FILE FOR TURBOFF PROGRAM

Generated by Turbdes Program

Effect of Variable Geometry
Turbofan-Turbojet Engine
Low pressure Turbine
Design point is The Subsonic Point (Altitude 9150 m, Mach Number 0.95).

```
LP TURBINE DESIGN            STAGE       1
 1.4790000E-02
  958.0000
  247496.0
  102.7000
 -3800.000
     4
 -3800.000
 -3420.000
 -3040.000
 -2660.000
  66.02087     68.11556     68.63924     70.63080
  46.21461     44.11991     43.59624     41.60468
 -5                                                    Blade Angle
 0.1091413
 0.1041582
 3.6614E-02                             Blade Opening
 0.3471941
 0.2094514
 1.7629892E-03
 0.2190096
  37.45944
 9.0181008E-02
 8.0462620E-02
 3.5435684E-02
 0.2682087
 0.2242110
 1.5671015E-03
 2.5000001E-04
 0.2703456
 0.4700000
     0
     1
  70.63080     73.66656     74.42552     77.59114
  41.60468     38.56891     37.80997     34.64434
  35.00001
 0.1559404
 0.1388438
 6.0033977E-02
 0.4628126
 0.2248738
 2.3506521E-03
 0.3206188
  37.27955
 0.1412314
 0.1247474
 5.5969652E-02
 0.4158245
 0.2263262
 2.4317091E-03
 2.5000001E-04
 0.3978117
 0.4700000
     0
     0
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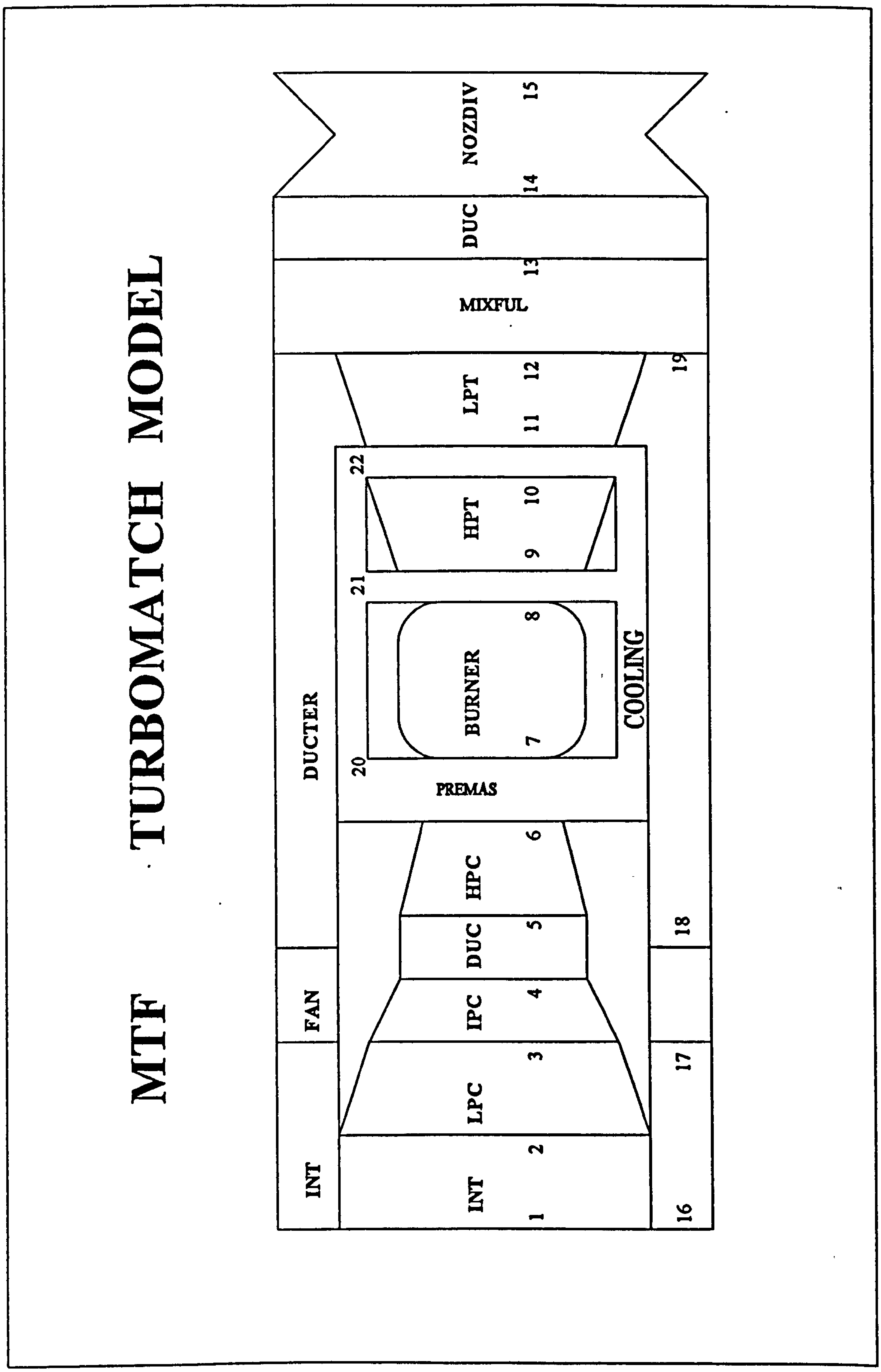
APPENDIX C

MID-TANDEM FAN ENGINE


MTF TURBOMATCH MODEL
INT
FAN
DUCTER
INT
LPC
IPC
DUC
HPC
PREMAS
BURNER
COOLING
HPT
LPT
MIXFUL
DUC
NOZDIV
1
2
3
4
5
6
7
8
9
10
11
12
13
14
15
16
17
18
19
20
21
22

TURBOMATCH INPUT FILE

Mid-Tandem Fan Engine

```
mtf.TAKE-OFF////
OP SI KE VA FP
 3
0 10 15 20 25 30
 0.50 0.61 0.71 0.76 0.81 0.86 0.91 0.96 1.01 1.111
 0.89 162. 0.351 1.03 156. 0.575 1.11 140. 0.885 1.19 117. 0.859 1.21  89. 0.681
 0.97 177. 0.188 1.15 172. 0.873 1.22 157. 0.905 1.29 135. 0.833 1.32 109. 0.686
 1.16 196. 0.559 1.29 191. 0.906 1.36 177. 0.893 1.43 156. 0.817 1.45 131. 0.695
 1.06 207. 0.188 1.37 202. 0.906 1.44 188. 0.885 1.51 167. 0.811 1.53 143. 0.700
 1.22 218. 0.488 1.46 214. 0.902 1.53 200. 0.878 1.60 180. 0.807 1.61 156. 0.704
 1.11 231. 0.218 1.55 226. 0.898 1.63 213. 0.873 1.70 193. 0.804 1.71 170. 0.708
 1.41 243. 0.633 1.65 239. 0.895 1.74 226. 0.869 1.81 208. 0.803 1.83 186. 0.712
 1.60 256. 0.525 1.76 252. 0.892 1.86 241. 0.868 1.94 223. 0.804 1.95 203. 0.719
 1.52 269. 0.100 1.88 266. 0.890 1.99 256. 0.870 2.07 241. 0.815 2.10 224. 0.740
 1.78 287. 0.926 2.18 286. 0.866 2.28 284. 0.860 2.35 279. 0.841 2.40 274. 0.817
 0.66 153. 0.157 0.98 148. 0.452 1.07 131. 0.847 1.17 107. 0.855 1.20  78. 0.648
 0.94 167. 0.894 1.08 162. 0.790 1.18 146. 0.903 1.27 123. 0.826 1.30  95. 0.652
 0.90 184. 0.559 1.21 178. 0.894 1.31 163. 0.897 1.39 140. 0.807 1.42 113. 0.660
 1.14 193. 0.566 1.28 188. 0.904 1.38 172. 0.891 1.47 150. 0.800 1.50 123. 0.664
 1.12 203. 0.365 1.35 197. 0.907 1.46 183. 0.885 1.55 160. 0.796 1.58 134. 0.668
 0.92 213. 0.214 1.44 208. 0.906 1.55 193. 0.880 1.64 171. 0.792 1.67 145. 0.672
 0.79 224. 0.514 1.52 219. 0.904 1.65 204. 0.876 1.74 183. 0.790 1.77 157. 0.676
 1.25 235. 0.441 1.62 230. 0.903 1.76 216. 0.874 1.86 195. 0.789 1.89 171. 0.680
 0.61 246. 0.798 1.72 241. 0.903 1.87 228. 0.875 1.99 209. 0.794 2.02 186. 0.690
 1.17 266. 0.223 1.95 263. 0.896 2.14 255. 0.882 2.27 243. 0.828 2.33 228. 0.757
 0.59 149. 0.108 0.95 143. 0.464 1.05 127. 0.802 1.16 102. 0.853 1.20  73. 0.631
 0.87 162. 0.443 1.05 156. 0.670 1.15 140. 0.899 1.26 117. 0.822 1.29  89. 0.635
 0.96 177. 0.289 1.16 172. 0.877 1.28 156. 0.898 1.38 133. 0.801 1.41 106. 0.642
 0.93 186. 0.391 1.23 180. 0.896 1.35 165. 0.892 1.45 142. 0.794 1.48 115. 0.646
 0.91 195. 0.363 1.30 190. 0.903 1.42 174. 0.886 1.53 151. 0.789 1.56 124. 0.650
 1.03 204. 0.100 1.38 199. 0.905 1.51 184. 0.880 1.62 161. 0.784 1.65 134. 0.653
 1.12 214. 0.308 1.46 209. 0.904 1.60 194. 0.876 1.71 172. 0.781 1.75 145. 0.656
 1.25 224. 0.518 1.55 219. 0.904 1.70 204. 0.873 1.82 183. 0.779 1.86 157. 0.659
 0.61 234. 0.897 1.64 229. 0.903 1.82 215. 0.872 1.95 195. 0.780 1.99 170. 0.664
 1.08 253. 0.128 1.85 249. 0.900 2.07 239. 0.880 2.23 224. 0.807 2.28 206. 0.715
 0.66 144. 0.441 0.92 139. 1.721 1.03 122. 0.707 1.15  97. 0.851 1.19  68. 0.614
 0.66 156. 0.530 1.01 151. 0.264 1.13 135. 0.891 1.24 111. 0.818 1.28  83. 0.618
 0.80 171. 0.230 1.12 165. 0.842 1.25 149. 0.897 1.36 126. 0.795 1.40  98. 0.624
 0.93 179. 0.459 1.18 173. 0.880 1.31 157. 0.892 1.43 134. 0.787 1.47 106. 0.628
 0.71 187. 0.161 1.25 181. 0.895 1.39 166. 0.886 1.50 143. 0.781 1.54 115. 0.631
 0.77 195. 0.970 1.32 190. 0.900 1.47 175. 0.880 1.59 151. 0.775 1.63 124. 0.633
 1.10 204. 0.293 1.39 199. 0.901 1.55 184. 0.874 1.68 161. 0.770 1.72 134. 0.635
 0.80 213. 0.581 1.47 208. 0.901 1.65 193. 0.870 1.79 171. 0.766 1.83 145. 0.637
 1.26 221. 0.552 1.56 216. 0.900 1.76 202. 0.867 1.91 181. 0.764 1.95 156. 0.639
 1.29 239. 0.478 1.76 235. 0.898 2.00 223. 0.874 2.17 206. 0.784 2.22 187. 0.676
 0.64 139. 0.355 0.88 134. 0.157 1.01 117. 0.390 1.14  93. 0.849 1.19  64. 0.596
 0.49 151. 0.135 0.97 145. 0.840 1.10 129. 0.875 1.23 105. 0.813 1.28  77. 0.599
 0.90 164. 0.290 1.07 158. 0.763 1.21 142. 0.895 1.34 119. 0.788 1.39  91. 0.605
 0.77 171. 0.272 1.13 165. 0.845 1.28 150. 0.891 1.41 126. 0.779 1.45  99. 0.607
 0.98 178. 0.131 1.19 173. 0.877 1.35 157. 0.884 1.48 134. 0.771 1.53 106. 0.610
 0.98 186. 0.095 1.26 181. 0.889 1.42 165. 0.877 1.56 142. 0.764 1.61 115. 0.611
 1.00 193. 0.261 1.33 188. 0.894 1.51 173. 0.870 1.65 150. 0.757 1.70 123. 0.612
 1.05 201. 0.155 1.40 196. 0.894 1.60 181. 0.864 1.75 159. 0.750 1.80 133. 0.611
 1.20 208. 0.520 1.48 203. 0.894 1.70 189. 0.859 1.86 168. 0.745 1.91 143. 0.612
 1.17 224. 0.352 1.67 219. 0.891 1.94 207. 0.861 2.11 189. 0.755 2.16 168. 0.636
 0.54 134. 0.549 0.85 129. 0.141 0.98 112. 0.261 1.13  88. 0.846 1.18  59. 0.574
 0.59 144. 0.562 0.93 139. 0.178 1.07 123. 0.841 1.22  99. 0.807 1.27  72. 0.578
 0.53 156. 0.609 1.02 151. 0.482 1.18 135. 0.889 1.32 112. 0.780 1.37  84. 0.583
 0.88 163. 0.508 1.07 157. 0.760 1.24 142. 0.887 1.39 119. 0.769 1.44  91. 0.585
 0.18 170. 0.161 1.13 164. 0.834 1.30 149. 0.881 1.46 126. 0.760 1.51  98. 0.586
 1.10 176. 0.710 1.20 170. 0.868 1.38 155. 0.872 1.53 132. 0.749 1.59 106. 0.586
 1.16 182. 0.790 1.27 177. 0.880 1.46 162. 0.863 1.62 140. 0.739 1.67 113. 0.584
 1.21 188. 0.817 1.34 183. 0.883 1.55 169. 0.854 1.72 147. 0.729 1.77 122. 0.581
 1.25 194. 0.822 1.41 190. 0.882 1.65 176. 0.847 1.82 155. 0.722 1.87 131. 0.580
 1.36 207. 0.815 1.58 203. 0.877 1.87 191. 0.843 2.04 173. 0.725 2.09 152. 0.597
 2
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  1.33 107. 0.760 1.38 105. 0.805 1.44  99. 0.870 1.50  92. 0.906 1.53  82. 0.879
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  1.61 124. 0.859 1.68 123. 0.877 1.73 119. 0.894 1.77 113. 0.896 1.79 106. 0.875
  1.69 129. 0.863 1.80 127. 0.883 1.85 124. 0.892 1.88 119. 0.889 1.89 113. 0.871
  1.77 134. 0.842 2.03 134. 0.874 2.09 133. 0.875 2.11 131. 0.871 2.13 129. 0.864
  0.87 77. 0.259 0.92 74. 0.123 1.02 67. 0.236 1.13 57. 0.837 1.18 44. 0.847
```

```
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1.34 107. 0.777 1.40 105. 0.823 1.48 100. 0.887 1.55 92. 0.916 1.58 83. 0.880
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1.53 118. 0.848 1.60 117. 0.874 1.67 112. 0.903 1.73 105. 0.910 1.75 96. 0.879
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1.70 130. 0.857 1.96 129. 0.891 2.01 127. 0.894 2.05 124. 0.890 2.06 121. 0.878
0.83 74. 0.155 0.88 72. 0.391 1.00 65. 0.566 1.12 54. 0.833 1.16 41. 0.832
0.93 80. 0.869 0.98 78. 0.211 1.09 71. 0.572 1.20 62. 0.888 1.24 50. 0.842
1.04 88. 0.247 1.10 86. 0.471 1.20 79. 0.772 1.30 70. 0.908 1.34 60. 0.853
1.11 92. 0.477 1.16 90. 0.623 1.26 84. 0.823 1.36 75. 0.913 1.40 65. 0.858
1.18 96. 0.622 1.24 94. 0.721 1.33 89. 0.857 1.43 80. 0.916 1.46 70. 0.864
1.26 101. 0.717 1.32  99. 0.786 1.41  94. 0.880 1.50  85. 0.917 1.53  76. 0.868
1.34 106. 0.780 1.40 104. 0.830 1.49  99. 0.895 1.58  91. 0.916 1.61  82. 0.870
1.43 111. 0.820 1.50 110. 0.858 1.58 104. 0.904 1.66  97. 0.914 1.69  88. 0.870
1.52 117. 0.847 1.60 115. 0.876 1.68 110. 0.908 1.76 103. 0.911 1.79  94. 0.869
1.65 126. 0.864 1.83 124. 0.896 1.90 120. 0.907 1.96 115. 0.900 1.99 108. 0.871
0.83 74. 0.155 0.88 72. 0.391 1.00 65. 0.566 1.12 54. 0.833 1.16 41. 0.832
0.93 80. 0.869 0.98 78. 0.211 1.09 71. 0.572 1.20 62. 0.888 1.24 50. 0.842
1.04 88. 0.247 1.10 86. 0.471 1.20 79. 0.772 1.30 70. 0.908 1.34 60. 0.853
1.11 92. 0.477 1.16 90. 0.623 1.26 84. 0.823 1.36 75. 0.913 1.40 65. 0.858
1.18 96. 0.622 1.24 94. 0.721 1.33 89. 0.857 1.43 80. 0.916 1.46 70. 0.864
1.26 101. 0.717 1.32  99. 0.786 1.41  94. 0.880 1.50  85. 0.917 1.53  76. 0.868
1.34 106. 0.780 1.40 104. 0.830 1.49  99. 0.895 1.58  91. 0.916 1.61  82. 0.870
1.43 111. 0.820 1.50 110. 0.858 1.58 104. 0.904 1.66  97. 0.914 1.69  88. 0.870
1.52 117. 0.847 1.60 115. 0.876 1.68 110. 0.908 1.76 103. 0.911 1.79  94. 0.869
1.65 126. 0.864 1.83 124. 0.896 1.90 120. 0.907 1.96 115. 0.900 1.99 108. 0.871
0.83 74. 0.155 0.88 72. 0.391 1.00 65. 0.566 1.12 54. 0.833 1.16 41. 0.832
0.93 80. 0.869 0.98 78. 0.211 1.09 71. 0.572 1.20 62. 0.888 1.24 50. 0.842
1.04 88. 0.247 1.10 86. 0.471 1.20 79. 0.772 1.30 70. 0.908 1.34 60. 0.853
1.11 92. 0.477 1.16 90. 0.623 1.26 84. 0.823 1.36 75. 0.913 1.40 65. 0.858
1.18 96. 0.622 1.24 94. 0.721 1.33 89. 0.857 1.43 80. 0.916 1.46 70. 0.864
1.26 101. 0.717 1.32  99. 0.786 1.41  94. 0.880 1.50  85. 0.917 1.53  76. 0.868
1.34 106. 0.780 1.40 104. 0.830 1.49  99. 0.895 1.58  91. 0.916 1.61  82. 0.870
1.43 111. 0.820 1.50 110. 0.858 1.58 104. 0.904 1.66  97. 0.914 1.69  88. 0.870
1.52 117. 0.847 1.60 115. 0.876 1.68 110. 0.908 1.76 103. 0.911 1.79  94. 0.869
1.65 126. 0.864 1.83 124. 0.896 1.90 120. 0.907 1.96 115. 0.900 1.99 108. 0.871
0.83 74. 0.155 0.88 72. 0.391 1.00 65. 0.566 1.12 54. 0.833 1.16 41. 0.832
0.93 80. 0.869 0.98 78. 0.211 1.09 71. 0.572 1.20 62. 0.888 1.24 50. 0.842
1.04 88. 0.247 1.10 86. 0.471 1.20 79. 0.772 1.30 70. 0.908 1.34 60. 0.853
1.11 92. 0.477 1.16 90. 0.623 1.26 84. 0.823 1.36 75. 0.913 1.40 65. 0.858
1.18 96. 0.622 1.24 94. 0.721 1.33 89. 0.857 1.43 80. 0.916 1.46 70. 0.864
1.26 101. 0.717 1.32  99. 0.786 1.41  94. 0.880 1.50  85. 0.917 1.53  76. 0.868
1.34 106. 0.780 1.40 104. 0.830 1.49  99. 0.895 1.58  91. 0.916 1.61  82. 0.870
1.43 111. 0.820 1.50 110. 0.858 1.58 104. 0.904 1.66  97. 0.914 1.69  88. 0.870
1.52 117. 0.847 1.60 115. 0.876 1.68 110. 0.908 1.76 103. 0.911 1.79  94. 0.869
1.65 126. 0.864 1.83 124. 0.896 1.90 120. 0.907 1.96 115. 0.900 1.99 108. 0.871
4
-5 0 5 5 5 5
0.50 0.60 0.70 0.75 0.80 0.85 0.90 0.95 1.00 1.10
1.05 38. 0.477 1.09 37. 0.654 1.19 34. 0.835 1.27 29. 0.841 1.30 23. 0.738
1.22 43. 0.798 1.27 42. 0.839 1.35 39. 0.870 1.43 34. 0.836 1.45 28. 0.748
1.41 49. 0.863 1.49 48. 0.880 1.57 45. 0.877 1.64 40. 0.836 1.66 35. 0.761
1.52 52. 0.875 1.61 51. 0.887 1.70 48. 0.878 1.76 44. 0.837 1.79 39. 0.768
1.63 56. 0.880 1.75 55. 0.891 1.84 52. 0.879 1.91 48. 0.839 1.93 43. 0.776
1.77 60. 0.884 1.91 59. 0.893 2.00 56. 0.881 2.07 52. 0.843 2.10 47. 0.784
1.90 64. 0.885 2.08 64. 0.895 2.19 61. 0.882 2.27 57. 0.847 2.29 52. 0.793
2.04 69. 0.885 2.27 68. 0.895 2.40 66. 0.884 2.49 62. 0.852 2.52 58. 0.802
2.18 73. 0.881 2.49 72. 0.893 2.64 70. 0.883 2.75 67. 0.853 2.78 63. 0.808
2.33 78. 0.846 3.00 78. 0.869 3.19 77. 0.859 3.29 76. 0.848 3.34 76. 0.837
1.05 38. 0.477 1.09 37. 0.654 1.19 34. 0.835 1.27 29. 0.841 1.30 23. 0.738
1.22 43. 0.798 1.27 42. 0.839 1.35 39. 0.870 1.43 34. 0.836 1.45 28. 0.748
1.41 49. 0.863 1.49 48. 0.880 1.57 45. 0.877 1.64 40. 0.836 1.66 35. 0.761
1.52 52. 0.875 1.61 51. 0.887 1.70 48. 0.878 1.76 44. 0.837 1.79 39. 0.768
1.63 56. 0.880 1.75 55. 0.891 1.84 52. 0.879 1.91 48. 0.839 1.93 43. 0.776
1.77 60. 0.884 1.91 59. 0.893 2.00 56. 0.881 2.07 52. 0.843 2.10 47. 0.784
1.90 64. 0.885 2.08 64. 0.895 2.19 61. 0.882 2.27 57. 0.847 2.29 52. 0.793
2.04 69. 0.885 2.27 68. 0.895 2.40 66. 0.884 2.49 62. 0.852 2.52 58. 0.802
2.18 73. 0.881 2.49 72. 0.893 2.64 70. 0.883 2.75 67. 0.853 2.78 63. 0.808
2.33 78. 0.846 3.00 78. 0.869 3.19 77. 0.859 3.29 76. 0.848 3.34 76. 0.837
1.05 38. 0.477 1.09 37. 0.654 1.19 34. 0.835 1.27 29. 0.841 1.30 23. 0.738
1.22 43. 0.798 1.27 42. 0.839 1.35 39. 0.870 1.43 34. 0.836 1.45 28. 0.748
1.41 49. 0.863 1.49 48. 0.880 1.57 45. 0.877 1.64 40. 0.836 1.66 35. 0.761
1.52 52. 0.875 1.61 51. 0.887 1.70 48. 0.878 1.76 44. 0.837 1.79 39. 0.768
1.63 56. 0.880 1.75 55. 0.891 1.84 52. 0.879 1.91 48. 0.839 1.93 43. 0.776
1.77 60. 0.884 1.91 59. 0.893 2.00 56. 0.881 2.07 52. 0.843 2.10 47. 0.784
1.90 64. 0.885 2.08 64. 0.895 2.19 61. 0.882 2.27 57. 0.847 2.29 52. 0.793
2.04 69. 0.885 2.27 68. 0.895 2.40 66. 0.884 2.49 62. 0.852 2.52 58. 0.802
2.18 73. 0.881 2.49 72. 0.893 2.64 70. 0.883 2.75 67. 0.853 2.78 63. 0.808
2.33 78. 0.846 3.00 78. 0.869 3.19 77. 0.859 3.29 76. 0.848 3.34 76. 0.837
1.05 38. 0.477 1.09 37. 0.654 1.19 34. 0.835 1.27 29. 0.841 1.30 23. 0.738
1.22 43. 0.798 1.27 42. 0.839 1.35 39. 0.870 1.43 34. 0.836 1.45 28. 0.748
1.41 49. 0.863 1.49 48. 0.880 1.57 45. 0.877 1.64 40. 0.836 1.66 35. 0.761
1.52 52. 0.875 1.61 51. 0.887 1.70 48. 0.878 1.76 44. 0.837 1.79 39. 0.768
1.63 56. 0.880 1.75 55. 0.891 1.84 52. 0.879 1.91 48. 0.839 1.93 43. 0.776
1.77 60. 0.884 1.91 59. 0.893 2.00 56. 0.881 2.07 52. 0.843 2.10 47. 0.784
1.90 64. 0.885 2.08 64. 0.895 2.19 61. 0.882 2.27 57. 0.847 2.29 52. 0.793
2.04 69. 0.885 2.27 68. 0.895 2.40 66. 0.884 2.49 62. 0.852 2.52 58. 0.802
2.18 73. 0.881 2.49 72. 0.893 2.64 70. 0.883 2.75 67. 0.853 2.78 63. 0.808
2.33 78. 0.846 3.00 78. 0.869 3.19 77. 0.859 3.29 76. 0.848 3.34 76. 0.837
1.05 38. 0.477 1.09 37. 0.654 1.19 34. 0.835 1.27 29. 0.841 1.30 23. 0.738
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1.22 43. 0.798 1.27 42. 0.839 1.35 39. 0.870 1.43 34. 0.836 1.45 28. 0.748
1.41 49. 0.863 1.49 48. 0.880 1.57 45. 0.877 1.64 40. 0.836 1.66 35. 0.761
1.52 52. 0.875 1.61 51. 0.887 1.70 48. 0.878 1.76 44. 0.837 1.79 39. 0.768
1.63 56. 0.880 1.75 55. 0.891 1.84 52. 0.879 1.91 48. 0.839 1.93 43. 0.776
1.77 60. 0.884 1.91 59. 0.893 2.00 56. 0.881 2.07 52. 0.843 2.10 47. 0.784
1.90 64. 0.885 2.08 64. 0.895 2.19 61. 0.882 2.27 57. 0.847 2.29 52. 0.793
2.04 69. 0.885 2.27 68. 0.895 2.40 66. 0.884 2.49 62. 0.852 2.52 58. 0.802
2.18 73. 0.881 2.49 72. 0.893 2.64 70. 0.883 2.75 67. 0.853 2.78 63. 0.808
2.33 78. 0.846 3.00 78. 0.869 3.19 77. 0.859 3.29 76. 0.848 3.34 76. 0.837
1.05 38. 0.477 1.09 37. 0.654 1.19 34. 0.835 1.27 29. 0.841 1.30 23. 0.738
1.22 43. 0.798 1.27 42. 0.839 1.35 39. 0.870 1.43 34. 0.836 1.45 28. 0.748
1.41 49. 0.863 1.49 48. 0.880 1.57 45. 0.877 1.64 40. 0.836 1.66 35. 0.761
1.52 52. 0.875 1.61 51. 0.887 1.70 48. 0.878 1.76 44. 0.837 1.79 39. 0.768
1.63 56. 0.880 1.75 55. 0.891 1.84 52. 0.879 1.91 48. 0.839 1.93 43. 0.776
1.77 60. 0.884 1.91 59. 0.893 2.00 56. 0.881 2.07 52. 0.843 2.10 47. 0.784
1.90 64. 0.885 2.08 64. 0.895 2.19 61. 0.882 2.27 57. 0.847 2.29 52. 0.793
2.04 69. 0.885 2.27 68. 0.895 2.40 66. 0.884 2.49 62. 0.852 2.52 58. 0.802
2.18 73. 0.881 2.49 72. 0.893 2.64 70. 0.883 2.75 67. 0.853 2.78 63. 0.808
2.33 78. 0.846 3.00 78. 0.869 3.19 77. 0.859 3.29 76. 0.848 3.34 76. 0.837
1
-10 -5 0 5 10 10
0.53 0.63 0.74 0.79 0.84 0.89 0.95 1.00 1.05 1.16
1.07 137. 0.161 1.15 134. 0.830 1.18 125. 0.898 1.21 111. 0.907 1.23  95. 0.835
1.00 152. 0.198 1.28 149. 0.900 1.31 142. 0.912 1.33 131. 0.894 1.34 118. 0.842
1.25 168. 0.561 1.43 166. 0.894 1.46 160. 0.897 1.47 152. 0.878 1.48 142. 0.841
1.28 176. 0.525 1.52 174. 0.886 1.54 169. 0.889 1.56 162. 0.870 1.56 153. 0.837
1.05 185. 0.250 1.61 184. 0.879 1.64 180. 0.884 1.66 173. 0.868 1.66 166. 0.838
1.08 195. 0.269 1.71 194. 0.875 1.75 190. 0.882 1.77 185. 0.871 1.78 179. 0.847
1.30 205. 0.369 1.83 203. 0.873 1.88 200. 0.881 1.90 196. 0.873 1.91 190. 0.853
1.23 213. 0.267 1.95 212. 0.864 2.02 209. 0.874 2.06 205. 0.870 2.07 200. 0.854
1.01 218. 0.056 2.07 218. 0.844 2.16 216. 0.859 2.21 214. 0.860 2.23 211. 0.853
1.10 226. 0.478 2.29 226. 0.776 2.42 225. 0.799 2.49 224. 0.802 2.52 223. 0.796
1.01 133. 0.120 1.11 130. 0.772 1.15 120. 0.885 1.20 105. 0.906 1.21  87. 0.819
1.06 147. 0.238 1.24 144. 0.890 1.27 135. 0.912 1.30 122. 0.894 1.31 107. 0.826
1.10 162. 0.192 1.38 159. 0.902 1.41 152. 0.905 1.44 141. 0.881 1.45 129. 0.829
1.28 169. 0.598 1.46 167. 0.899 1.49 161. 0.898 1.52 151. 0.875 1.53 140. 0.828
1.18 177. 0.343 1.55 175. 0.894 1.58 170. 0.893 1.61 161. 0.870 1.62 151. 0.827
1.19 186. 0.314 1.64 184. 0.890 1.68 179. 0.891 1.71 172. 0.871 1.72 163. 0.833
1.12 195. 0.181 1.74 193. 0.887 1.80 189. 0.890 1.83 183. 0.875 1.84 175. 0.843
1.16 203. 0.215 1.86 202. 0.880 1.93 198. 0.888 1.97 193. 0.878 1.98 186. 0.851
1.39 210. 0.109 1.98 209. 0.869 2.07 206. 0.880 2.12 202. 0.876 2.14 197. 0.857
1.84 219. 0.710 2.20 219. 0.819 2.34 218. 0.840 2.41 217. 0.845 2.45 215. 0.841
1.09 129. 0.123 1.07 125. 0.667 1.13 115. 0.865 1.18  99. 0.905 1.20  80. 0.805
1.08 142. 0.143 1.19 138. 0.867 1.24 129. 0.910 1.28 115. 0.893 1.30  98. 0.811
1.10 155. 0.308 1.33 152. 0.901 1.37 144. 0.909 1.41 132. 0.881 1.42 118. 0.815
1.20 162. 0.495 1.40 160. 0.902 1.45 152. 0.904 1.48 141. 0.875 1.49 128. 0.815
1.24 170. 0.516 1.48 167. 0.900 1.53 160. 0.899 1.56 150. 0.870 1.58 138. 0.815
1.22 177. 0.408 1.57 175. 0.897 1.62 169. 0.895 1.66 159. 0.868 1.67 148. 0.817
1.20 185. 0.330 1.66 183. 0.894 1.73 178. 0.894 1.77 169. 0.871 1.78 160. 0.825
1.28 193. 0.398 1.77 191. 0.889 1.84 187. 0.894 1.89 180. 0.877 1.91 172. 0.838
1.12 200. 0.170 1.88 199. 0.881 1.97 195. 0.891 2.03 189. 0.879 2.05 182. 0.845
1.81 211. 0.764 2.12 210. 0.847 2.26 209. 0.866 2.33 206. 0.868 2.37 203. 0.859
1.10 124. 0.169 1.03 121. 0.430 1.10 110. 0.831 1.16  94. 0.903 1.19  74. 0.793
1.06 136. 0.167 1.14 133. 0.825 1.20 123. 0.903 1.26 108. 0.892 1.28  90. 0.798
1.00 149. 0.488 1.27 146. 0.892 1.33 137. 0.909 1.37 124. 0.880 1.39 108. 0.801
1.08 155. 0.738 1.34 152. 0.900 1.40 144. 0.906 1.44 131. 0.873 1.46 117. 0.802
1.12 162. 0.313 1.42 159. 0.901 1.48 151. 0.901 1.52 140. 0.868 1.54 126. 0.802
1.06 168. 0.139 1.50 166. 0.899 1.56 159. 0.897 1.61 148. 0.865 1.62 135. 0.803
1.00 175. 0.308 1.59 173. 0.896 1.66 167. 0.895 1.71 157. 0.865 1.72 145. 0.807
1.45 183. 0.712 1.68 181. 0.893 1.76 175. 0.895 1.82 166. 0.869 1.84 156. 0.817
1.62 190. 0.829 1.79 188. 0.887 1.88 183. 0.894 1.95 176. 0.875 1.97 167. 0.830
1.81 201. 0.821 2.02 200. 0.860 2.15 198. 0.880 2.24 193. 0.879 2.28 189. 0.857
0.50 120. 0.769 0.99 116. 0.440 1.07 105. 0.771 1.15  88. 0.902 1.17  69. 0.785
0.67 131. 0.379 1.09 127. 0.738 1.17 117. 0.890 1.24 102. 0.891 1.26  83. 0.788
0.97 143. 0.137 1.22 139. 0.870 1.28 130. 0.907 1.35 116. 0.878 1.37  99. 0.790
0.99 148. 0.042 1.28 145. 0.889 1.35 136. 0.906 1.41 123. 0.871 1.43 107. 0.790
1.22 154. 0.663 1.36 151. 0.896 1.43 142. 0.902 1.48 130. 0.865 1.50 115. 0.790
1.34 159. 0.840 1.43 157. 0.897 1.51 149. 0.897 1.56 137. 0.860 1.58 123. 0.789
1.41 165. 0.852 1.52 163. 0.895 1.59 156. 0.894 1.65 145. 0.857 1.67 132. 0.790
1.48 172. 0.858 1.61 169. 0.892 1.69 163. 0.893 1.76 153. 0.859 1.78 142. 0.797
1.57 178. 0.862 1.70 176. 0.888 1.80 171. 0.893 1.87 162. 0.865 1.89 152. 0.808
1.73 190. 0.833 1.91 189. 0.867 2.04 185. 0.886 2.14 179. 0.876 2.18 173. 0.839
0.50 120. 0.769 0.99 116. 0.440 1.07 105. 0.771 1.15  88. 0.902 1.17  69. 0.785
0.67 131. 0.379 1.09 127. 0.738 1.17 117. 0.890 1.24 102. 0.891 1.26  83. 0.788
0.97 143. 0.137 1.22 139. 0.870 1.28 130. 0.907 1.35 116. 0.878 1.37  99. 0.790
0.99 148. 0.042 1.28 145. 0.889 1.35 136. 0.906 1.41 123. 0.871 1.43 107. 0.790
1.22 154. 0.663 1.36 151. 0.896 1.43 142. 0.902 1.48 130. 0.865 1.50 115. 0.790
1.34 159. 0.840 1.43 157. 0.897 1.51 149. 0.897 1.56 137. 0.860 1.58 123. 0.789
1.41 165. 0.852 1.52 163. 0.895 1.59 156. 0.894 1.65 145. 0.857 1.67 132. 0.790
1.48 172. 0.858 1.61 169. 0.892 1.69 163. 0.893 1.76 153. 0.859 1.78 142. 0.797
1.57 178. 0.862 1.70 176. 0.888 1.80 171. 0.893 1.87 162. 0.865 1.89 152. 0.808
1.73 190. 0.833 1.91 189. 0.867 2.04 185. 0.886 2.14 179. 0.876 2.18 173. 0.839
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-1
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-1
INTAKE S1-2           D1-4            R200
COMPRE S2-3           D9-15    R202   V9      V10
ARITHY                D76-80
ARITHY                D95-99
ARITHY              D111-115
ARITHY              D116-120
COMPRE S3-4           D88-94   R203   V88
DUCTER S4-5         D5-8
INTAKE S16-17       D61-64      R199
COMPRE S17-18         D69-75   R204   V69
DUCTER S18,19         D65-68
ARITHY                D81-87
ARITHY        D100-106
COMPRE S5-6           D16-22   R205   V16     V17
PREMAS S6,7,20        D23-26
PREMAS S20,21,22    D27-30
BURNER S7-8           D31-33   R206   L1
MIXEES S8,21,9
TURBIN S9-10          D34-41,205,42   V35
MIXEES S10,22,11
TURBIN S11-12         D43-50,211,51   V44
MIXFUL S12,19,13    D52-54
DUCTER S13,14         D55-58   R207
NOZDIV S14,15,1     D59-60     R208
ARITHY              D250-256
ARITHY              D257-263
ARITHY              D264-270
ARITHY              D273-279
ARITHY              D280-288
ARITHY              D289-297
ARITHY              D298-306
ARITHY              D307-315
PLOTBD              D450,350,451,351,452,352,453,353
PERFOR S1,16,0        D107-110,208,200,206,0,199,207,0,0,0
CODEND

DATA////
1 0
2 0
3 0
4 -1

5 0.0
6 0.0
7 0.0
8 100000

9 0.885
10 1.005
11 1.86
12 0.885
13 0
14 1
15 0

16 0.8
17 0.975
18 2.54
19 0.88
20 1
21 4
22 0.0

23 0.85
24 0
25 1
26 0.01

27 0.65
28 0
29 1
30 0.01

31 0.05
32 0.99
33 -1.0

34 100000.0
35 -1
36 -1
37 0.9
38 -1
39 4
40 4
41 -1
42 0

43 0
44 -1
45 -1
46 0.9
47 -1
```

```
48 1
49 4
50 -1
51 0.0

52 3
53 1
54 0.42

55 1
56 0.03
57 0.91
58 100000

59 -1
60 -1

61 0
62 0.0
63 0
64 -1

65 0
66 0
67 0
68 100000

69 0.54
70 1.0
71 1.83
72 0.85
73 0
74 3
75 0

76 5
77 -1
78 70
79 -1
80 10

81 1
82 -1
83 210
84 -1
85 202
86 -1
87 203

88 0.78
89 0.9750
90 1.8
91 0.895
92 1
93 2
94 0

95 5
96 -1
97 89
98 -1
99 10

100 1
101 -1
102 211
103 -1
104 210
105 -1
106 204

107 -1
108 -1
109 0
110 82500.0

111 5
112 -1
113 61
114 -1
115 1

116 5
117 -1
118 63
119 -1
120 3

247 0.058910
248 288.15
249 1

250 4
```

```
251 -1
252 350    !pr lpc
253 3
254 4
255 2
256 4

257 4
258 -1
259 351     !pr ipc
260 4
261 4
262 3
263 4

264 4
265 -1
266 352   ! pr fan
267 18
268 4
269 17
270 4

273 4
274 -1
275 353   !PR COM 1
276 6
277 4
278 5
279 4

280 29
281 -1
282 450
283 2    !ndmf lpc
284 2
285 2
286 6
287 2
288 4

289 29
290 -1
291 451
292 3   !ndmf  ipc
293 2
294 3
295 6
296 3
297 4

298 29
299 -1
300 452
301 17
302 2  !ndmf  fan
303 17
304 6
305 17
306 4

307 29
308 -1
309 453
310 5
311 2  !ndmf  hpc
312 5
313 6
314 5
315 4

-1
1 2 188
16 2 266
8 6 1230
-1
59 -2
1 460
3 0.5
-1
8 6 1450
-1
1 2290
3 0.6
-1
8 6 1355
-1
60 1.43
1 9150
3 0.95
-1
8 6 1150
```

```
-1
-1
75 5
59 -1
60 1.85
55 2
-1
8 6 1220
14 6 1100
15 8 2.2
12 8 1
19 8 0.7
-1
75 10
1 9750
3 1.05
-1
8 6 1240
14 6 1070
-1
75 15
60 1.67
-1
60 1.5
75 20
1 12250
3 1.6
-1
8 6 1390
14 6 1160
-1
60 1.5
75 25
-1
8 6 1300
-1
55 1
59 -1
60 1.15
75 30
-1
-1
75 30
-1
-1
75 30
55 2
15 5
60 1.21
1 18750
3 2.7
-1
8 6 1800
19 8 0.43
12 8 1.0
15 8 2.4
14 6 1230
-1
60 1.12
94 -4
55 1
15 2
-1
19 8 0.36
-1
-3
```

TURBOMATCH OUTPUT FILE

```
Mid-Tandem Fan Engine


  TURBOMATCH SCHEME-VAX VERSION (11/11/94)

 LIMITS:100 Codewords, 800 Brick Data Items, 50 Station Vector
 15 BD Items printable by any call of:-
 OUTPUT, OUTPBD, OUTPSV, PLOTIT, PLOTBD or PLOTSV


      ***** DESIGN POINT ENGINE CALCULATIONS *****


 ***** AMBIENT AND INLET PARAMETERS *****
 Alt. =      0.0      I.S.A. Dev. =    0.000       Mach No. =  0.00
 Etar = 1.0000        Momentum Drag =      0.00

 ***** COMPRESSOR  1 PARAMETERS *****
 PRSF =  0.10140E+01     ETASF =  0.99077E+00     WASF =  0.99976E+00
 Z = 0.88500             PR =   1.860             ETA = 0.88500
```

```
PCN =   1.0050          CN = 1.00500             COMWK =  0.11929E+08
STATOR ANGLE =   0.00

***** COMPRESSOR  2 PARAMETERS *****
PRSF =  0.10035E+01     ETASF =  0.99004E+00     WASF =  0.96545E+00
Z = 0.78000             PR =   1.800             ETA = 0.89500
PCN =   1.0050          CN = 1.00500             COMWK =  0.13548E+08
STATOR ANGLE =   0.00

***** AMBIENT AND INLET PARAMETERS *****
Alt. =       0.0       I.S.A. Dev. =    0.000     Mach No. =  0.00
Etar = 1.0000         Momentum Drag =       0.00

***** COMPRESSOR  3 PARAMETERS *****
PRSF =  0.10015E+01     ETASF =  0.10664E+01     WASF =  0.10052E+01
Z = 0.54000             PR =   1.830             ETA = 0.85000
PCN =   1.0050          CN = 1.00500             COMWK =  0.17072E+08
STATOR ANGLE =   0.00

***** COMPRESSOR  4 PARAMETERS *****
PRSF =  0.99870E+00     ETASF =  0.10040E+01     WASF =  0.10119E+01
Z = 0.80000             PR =   2.540             ETA = 0.88000
PCN =   0.9750          CN = 0.97500             COMWK =  0.27603E+08
STATOR ANGLE =   0.00

***** COMBUSTION CHAMBER PARAMETERS *****
ETASF =  0.99000E+00
ETA = 0.99000           DLP = 0.4257             WFB =   2.9825

***** TURBINE  1 PARAMETERS *****
CNSF =  0.72203E+02     ETASF =  0.10567E+01     TFSF =  0.54147E+00
DHSF =  0.76587E+04
TF =  414.346            ETA = 0.90000            CN =  2.060
AUXWK =  0.10000E+06     NGV ANGLE = 0.00

***** TURBINE  2 PARAMETERS *****
CNSF =  0.65358E+02     ETASF =  0.10567E+01     TFSF =  0.32049E+00
DHSF =  0.12814E+05
TF =  414.346            ETA = 0.90000            CN =  2.060
AUXWK =  0.00000E+00     NGV ANGLE = 0.00

***** MIXING MACH NUMBERS *****
Station 12, M= 0.421    Station 19,M = 0.426    Station 13,M = 0.442

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA =  0.9100           DLP = 0.0543             WFB =   0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
Throat Area =  1.5013    Throat Velocity =  431.23    Throat Mach No. =  1.0000
Exit Area =  1.5064      Exit Velocity =  408.29      Exit Mach No. =  0.9377
Nozzle Coeff. =  0.9796  Gross Thrust = 182783.84

Nozzle exit and throat areas are fixed.
AREA RATIO (EXIT/THROAT): 1.00


Scale Factor on above Mass Flows, Areas, Thrusts & Powers =     1.0000
```

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	188.000	1.00000	1.00000	288.15	288.15	0.0	******
2	0.00000	188.000	******	1.00000	******	288.15	******	******
3	0.00000	188.000	******	1.86000	******	351.28	******	******
4	0.00000	188.000	******	3.34800	******	422.52	******	******
5	0.00000	188.000	******	3.34800	******	422.52	******	******
6	0.00000	188.000	******	8.50392	******	565.29	******	******
7	0.00000	159.800	******	8.51392	******	565.29	******	******
8	0.01866	162.783	******	8.08822	******	1230.00	******	******
9	0.01674	181.113	******	8.08822	******	1167.85	******	******
10	0.01674	181.113	******	4.71029	******	1039.58	******	******
11	0.01586	190.983	******	4.71029	******	1016.70	******	******
12	0.01586	190.983	1.61647	1.81727	799.35	823.53	233.5	1.1456
13	0.00657	456.983	1.58571	1.80887	537.42	557.17	203.8	2.1527
14	0.00657	456.983	******	1.75461	******	557.17	******	******
15	0.00657	456.983	1.00000	1.75461	477.12	557.17	408.3	1.5064
16	0.00000	266.000	1.00000	1.00000	288.15	288.15	0.0	******
17	0.00000	266.000	******	1.00000	******	288.15	******	******
18	0.00000	266.000	******	1.83000	******	352.00	******	******
19	0.00000	266.000	1.61647	1.83000	339.73	352.00	157.2	1.0072
20	0.00000	28.200	******	8.50392	******	565.29	******	******
21	0.00000	18.330	******	8.51392	******	565.29	******	******
22	0.00000	9.870	******	8.50392	******	565.29	******	******

```
  Gross Thrust = 182783.84
Momentum Drag =       0.00
   Net Thrust = 182783.84
    Fuel Flow =     2.9825
       s.f.c. =  16.31726
   Sp. Thrust =    402.608
        Time Now 16:10:42

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```

```
60 1.43
1 9150
3 0.95
-1
8 6 1150
-1
                                                  Time Now 16:10:42

**********************************************

     ***** OFF DESIGN ENGINE CALCULATIONS. Converged after  10 Loops *****


***** AMBIENT AND INLET PARAMETERS *****
Alt. =   9150.0      I.S.A. Dev. =    0.000      Mach No. =  0.95
Etar = 1.0000        Momentum Drag =  28538.66

***** COMPRESSOR  1 PARAMETERS *****
PRSF =  0.10140E+01     ETASF =  0.99077E+00    WASF =  0.99976E+00
Z = 0.93316             PR =   1.842            ETA = 0.87724
PCN =   0.9538          CN = 0.98518            COMWK =  0.58424E+07
STATOR ANGLE =   0.00

***** COMPRESSOR  2 PARAMETERS *****
PRSF =  0.10035E+01     ETASF =  0.99004E+00    WASF =  0.96545E+00
Z = 0.79912             PR =   1.768            ETA = 0.89618
PCN =   0.9538          CN = 0.98580            COMWK =  0.64620E+07
STATOR ANGLE =   0.00

***** AMBIENT AND INLET PARAMETERS *****
Alt. =   9150.0      I.S.A. Dev. =    0.000      Mach No. =  0.95
Etar = 1.0000        Momentum Drag =  40639.30

***** COMPRESSOR  3 PARAMETERS *****
PRSF =  0.10015E+01     ETASF =  0.10664E+01    WASF =  0.10052E+01
Z = 0.61137             PR =   1.846            ETA = 0.93693
PCN =   0.9538          CN = 0.98518            COMWK =  0.78180E+07
STATOR ANGLE =   0.00

***** COMPRESSOR  4 PARAMETERS *****
PRSF =  0.99870E+00     ETASF =  0.10040E+01    WASF =  0.10119E+01
Z = 0.81075             PR =   2.507            ETA = 0.87784
PCN =   0.9329          CN = 0.96674            COMWK =  0.13375E+08
STATOR ANGLE =   0.00

***** COMBUSTION CHAMBER PARAMETERS *****
ETA = 0.99000           DLP = 0.2157            WFB =   1.4516

***** TURBINE  1 PARAMETERS *****
CNSF =  0.72203E+02     ETASF =  0.10567E+01    TFSF =  0.54147E+00
DHSF =  0.76587E+04
TF =  413.535            ETA = 0.89916           CN =  2.039
AUXWK =  0.10000E+06     NGV ANGLE = 0.00

***** TURBINE  2 PARAMETERS *****
CNSF =  0.65358E+02     ETASF =  0.10567E+01    TFSF =  0.32049E+00
DHSF =  0.12814E+05
TF =  411.161            ETA = 0.89947           CN =  2.023
AUXWK =  0.00000E+00     NGV ANGLE = 0.00

***** MIXING MACH NUMBERS *****
Station 12, M= 0.398    Station 19,M = 0.401    Station 13,M = 0.415

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA =  0.9100           DLP = 0.0290            WFB =   0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
Throat Area =  1.4300    Throat Velocity =  416.33    Throat Mach No. =  1.0000
Exit Area =  1.5932      Exit Velocity =  541.95      Exit Mach No. =  1.3977
Nozzle Coeff. =  0.9793  Gross Thrust = 128229.66

Nozzle exit area floating/throat area fixed.
AREA RATIO (EXIT/THROAT): 1.11


Scale Factor on above Mass Flows, Areas, Thrusts & Powers =     1.0000
```

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	99.052	0.29663	0.53043	228.68	270.10	288.1	******
2	0.00000	99.052	******	0.53043	******	270.10	******	******
3	0.00000	99.133	******	0.97706	******	328.85	******	******
4	0.00000	99.133	******	1.72779	******	393.47	******	******
5	0.00000	99.104	******	1.72779	******	393.47	******	******
6	0.00000	99.104	******	4.33098	******	525.49	******	******
7	0.00000	84.238	******	4.34098	******	525.49	******	******
8	0.01723	85.690	******	4.12530	******	1150.00	******	******
9	0.01546	95.353	******	4.12530	******	1091.41	******	******
10	0.01546	95.353	******	2.41581	******	971.27	******	******
11	0.01465	100.556	******	2.41581	******	949.69	******	******
12	0.01465	100.556	0.87338	0.96899	753.37	773.80	214.7	1.1456
13	0.00604	241.606	0.86115	0.96804	502.40	518.90	185.5	2.1527
14	0.00604	241.606	******	0.93900	******	518.90	******	******
15	0.00604	241.606	0.29663	0.93900	376.03	518.90	541.9	1.5932
16	0.00000	141.051	0.29663	0.53043	228.68	270.10	288.1	******
17	0.00000	141.051	******	0.53043	******	270.10	******	******
18	0.00000	141.051	******	0.97905	******	325.31	******	******
19	0.00000	141.051	0.87721	0.97905	315.27	325.31	142.7	1.0072
20	0.00000	14.866	******	4.33098	******	525.49	******	******
21	0.00000	9.663	******	4.34098	******	525.49	******	******
22	0.00000	5.203	******	4.33098	******	525.49	******	******

```
 Gross Thrust = 128229.66
Momentum Drag =  69177.96
   Net Thrust =  59051.70
    Fuel Flow =    1.4516
       s.f.c. =  24.58153
   Sp. Thrust =   245.944
         Time Now 16:10:43

***********************************************
60 1.12
 94 -4
 55 1
 15 2
 -1
 19 8 0.36
 -1
                                        Time Now 16:10:45

 ***********************************************

     ***** OFF DESIGN ENGINE CALCULATIONS. Converged after   5 Loops *****


 ***** AMBIENT AND INLET PARAMETERS *****
 Alt. =  18750.0       I.S.A. Dev. =    0.000      Mach No. =  2.70
 Etar = 0.8465         Momentum Drag = 130432.74

 ***** VARIABLE COMPRESSOR  1 PARAMETERS *****
 PRSF =  0.10140E+01      ETASF =  0.99077E+00     WASF =  0.99976E+00
 Z = 0.93933              PR =   1.736             ETA = 0.87676
 PCN =   1.2931           CN = 0.95319             COMWK =  0.16924E+08
 STATOR ANGLE =    2.00

 ***** VARIABLE COMPRESSOR  2 PARAMETERS *****
 PRSF =  0.10035E+01      ETASF =  0.99004E+00     WASF =  0.96545E+00
 Z = 0.93455              PR =   1.796             ETA = 0.88160
 PCN =   1.2931           CN = 0.96621             COMWK =  0.21230E+08
 STATOR ANGLE =  -4.00

 ***** AMBIENT AND INLET PARAMETERS *****
 Alt. =  18750.0       I.S.A. Dev. =    0.000      Mach No. =  2.70
 Etar = 0.8465         Momentum Drag =  93008.18

 ***** VARIABLE COMPRESSOR  3 PARAMETERS *****
 PRSF =  0.10015E+01      ETASF =  0.10664E+01     WASF =  0.10052E+01
 Z = 0.99900              PR =   1.757             ETA = 0.62177
 PCN =   1.2931           CN = 0.95319             COMWK =  0.17415E+08
 STATOR ANGLE =  30.00

 ***** COMPRESSOR  4 PARAMETERS *****
 PRSF =  0.99870E+00      ETASF =  0.10040E+01     WASF =  0.10119E+01
 Z = 0.64763              PR =   2.247             ETA = 0.89080
 PCN =   1.2315           CN = 0.92416             COMWK =  0.35646E+08
 STATOR ANGLE =    0.00

 ***** COMBUSTION CHAMBER PARAMETERS *****
 ETA = 0.99000            DLP = 0.5037             WFB =   3.7472

 ***** TURBINE  1 PARAMETERS *****
 CNSF =  0.72203E+02      ETASF =  0.10567E+01     TFSF =  0.54147E+00
 DHSF =  0.76587E+04
 TF =  413.675             ETA = 0.90504            CN =  2.144
 AUXWK =  0.10000E+06     NGV ANGLE = 0.00

 ***** TURBINE  2 PARAMETERS *****
 CNSF =  0.65358E+02      ETASF =  0.10567E+01     TFSF =  0.32049E+00
```

```
DHSF =  0.12814E+05
TF =  413.563              ETA = 0.90654           CN =  2.170
AUXWK =  0.00000E+00       NGV ANGLE = 0.00

***** MIXING MACH NUMBERS *****
Station 12, M= 0.504    Station 19,M = 0.685    Station 13,M = 0.566

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA =  0.9100             DLP = 0.0617            WFB =   0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
Throat Area =  1.1200    Throat Velocity =  580.77    Throat Mach No. =  1.0000
Exit Area =  2.4000      Exit Velocity = 1039.84      Exit Mach No. =  2.2529
Nozzle Coeff. =  0.9522  Gross Thrust = 306101.44

Nozzle exit and throat areas are fixed.
AREA RATIO (EXIT/THROAT): 2.14
Nozzle is Underexpanded.


Scale Factor on above Mass Flows, Areas, Thrusts & Powers =     1.0000
```

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	163.647	0.06577	1.53692	216.65	530.28	797.0	******
2	0.00000	163.647	******	1.30097	******	530.28	******	******
3	0.00000	163.403	******	2.25853	******	629.15	******	******
4	0.00000	163.403	******	4.05655	******	750.33	******	******
5	0.00000	163.593	******	4.05655	******	750.33	******	******
6	0.00000	163.593	******	9.11409	******	946.72	******	******
7	0.00000	139.054	******	9.12409	******	946.72	******	******
8	0.02695	142.802	******	8.62035	******	1800.00	******	******
9	0.02417	158.752	******	8.62035	******	1720.91	******	******
10	0.02417	158.752	******	5.05042	******	1545.10	******	******
11	0.02291	167.341	******	5.05042	******	1516.54	******	******
12	0.02291	167.341	1.68461	1.98266	1203.80	1250.59	338.7	1.0000
13	0.01337	284.033	1.67027	2.05650	973.99	1025.15	345.0	1.3600
14	0.01337	284.033	******	1.99480	******	1025.15	******	******
15	0.01337	284.033	0.16803	1.99480	540.39	1025.15	1039.8	2.4000
16	0.00000	116.692	0.06577	1.53692	216.65	530.28	797.0	******
17	0.00000	116.692	******	1.30097	******	530.28	******	******
18	0.00000	116.692	******	2.28573	******	672.28	******	******
19	0.00000	116.692	1.68014	2.28573	618.58	672.28	338.3	0.3600
20	0.00000	24.539	******	9.11409	******	946.72	******	******
21	0.00000	15.950	******	9.12409	******	946.72	******	******
22	0.00000	8.589	******	9.11409	******	946.72	******	******

```
 Gross Thrust = 306101.44
Momentum Drag = 223440.92
   Net Thrust =  82660.50
    Fuel Flow =    3.7472
       s.f.c. =  45.33295
   Sp. Thrust =   294.859
        Time Now 16:10:45

************************************************
 -3
```

INPUT FILE FOR THE L.P. COMPRESSOR

For Compdes Program

```
Mid-Tandem Fan Engine.
Design point is The Take-Off Point.


L.P COMPRESSOR DESIGN          !TITLE
2                                        !NOMBER OF STAGE
3                                        !ANNULUS GEOMETRY SELECTOR
1                                        !IGV's SELECTION
0.5                                      !HUB/TIP RATIO
88                                       !ISENTROPIC EFFICIENCY
180 180 200 200           !INLET AXIAL VELOCITY
30 33.48                       !TEMP. RISE PER STAGE
288.15                                   !INLET TOTAL TEMPERATURE
1                                        !INLET TOTAL PRESSURE
1.86                                     !DESIGN PRESSURE RATIO
188                                      !INLET MASS FLOW
1                                        !VORTEX DESIGN SELECTOR
0 20 20 0                      !STAGE INLET SWIRL ANGLE
5260                                     !ROTATIONAL SPEED
0.98 0.93                      !STAGE WORK DONE FACTORS
0.8 0.7 0.8 0.7                !S/C RATIO
3 2.75                         !ROTOR H/C RATIO
0.8 0.7                        !BMH REACTION
0.01 0.01                      !ROTOR TIP CLEARANCE
0.00001                        !EQUIVALENT SAND ROUGHNESS
```

INPUT FILE FOR THE FAN

For Compdes Program

```
Mid-Tandem Fan Engine.
Design point is The Take-Off Point.


FAN DESIGN                     !TITLE
2                                        !NOMBER OF STAGE
2                                        !ANNULUS GEOMETRY SELECTOR
1                                        !IGV's SELECTION
0.54                                     !HUB/TIP RATIO
88                                       !ISENTROPIC EFFICIENCY
180 180 180 180           !INLET AXIAL VELOCITY
30 33.48                       !TEMP. RISE PER STAGE
288.15                                   !INLET TOTAL TEMPERATURE
1                                        !INLET TOTAL PRESSURE
1.86                                     !DESIGN PRESSURE RATIO
266                                      !INLET MASS FLOW
1                                        !VORTEX DESIGN SELECTOR
15 15 15 15                    !STAGE INLET SWIRL ANGLE
5050                                     !ROTATIONAL SPEED
0.98 0.93                      !STAGE WORK DONE FACTORS
0.8 0.7 0.8 0.7                !S/C RATIO
3 2.75                         !ROTOR H/C RATIO
0.8 0.7                        !BMH REACTION
0.01 0.01                      !ROTOR TIP CLEARANCE
0.00001                        !EQUIVALENT SAND ROUGHNESS
```

INPUT FILE FOR THE L.P. COMPRESSOR

For Compdes Program

Mid-Tandem Fan Engine.
Design point is The Take-Off Point.

```
LP COMPRESSOR DESIGN                    !TITLE
4                                                       !NOMBER OF STAGE
2                                                       !ANNULUS GEOMETRY SELECTOR
1                                                       !IGV's SELECTION
0.35                                                    !HUB/TIP RATIO
88                                                      !ISENTROPIC EFFICIENCY
170 170 170 170 170 170     !INLET AXIAL VELOCITY
18.3 18.3 18.3 18.3         !TEMP. RISE PER STAGE
351.5                                                   !INLET TOTAL TEMPERATURE
1.86                                                    !INLET TOTAL PRESSURE
1.8                                                     !DESIGN PRESSURE RATIO
188.11                                  !INLET MASS FLOW
1                                                       !VORTEX DESIGN SELECTOR
0 0 0 0 0 0                             !STAGE INLET SWIRL ANGLE
5550                                                    !ROTATIONAL SPEED
0.98 0.95 0.92 0.9                      !STAGE WORK DONE FACTORS
0.8 0.7 0.8 0.7 0.8 0.7     0.8 0.7     !S/C RATIO
3.5 3.2 3 2.8                           !ROTOR H/C RATIO
0.8 0.7 0.7 0.7                         !BMH REACTION
0.01 0.01 0.01 0.01                     !ROTOR TIP CLEARANCE
0.00001                                 !EQUIVALENT SAND ROUGHNESS
```

INPUT FILE FOR THE H.P. COMPRESSOR

For Compdes Program

Mid-Tandem Fan Engine.
Design point is The Take-Off Point.

```
HP COMPRESSOR DESIGN                    !TITLE
3                                                       !NOMBER OF STAGE
3                                                       !ANNULUS GEOMETRY SELECTOR
0                                                       !IGV's SELECTION
0.7                                                     !HUB/TIP RATIO
88                                                      !ISENTROPIC EFFICIENCY
200 200 200 200 200         !INLET AXIAL VELOCITY
45.27 50 48                             !TEMP. RISE PER STAGE
424.15                                                  !INLET TOTAL TEMPERATURE
3.348                                                   !INLET TOTAL PRESSURE
2.54                                                    !DESIGN PRESSURE RATIO
188.11                                  !INLET MASS FLOW
1                                                       !VORTEX DESIGN SELECTOR
0 15 15 10 0                            !STAGE INLET SWIRL ANGLE
8200                                                    !ROTATIONAL SPEED
0.99 0.95 0.92                          !STAGE WORK DONE FACTORS
0.8 0.75 0.8 0.75 0.8 0.75  !S/C RATIO
2 2 2                                                   !ROTOR H/C RATIO
0.82 0.71 0.69                          !BMH REACTION
0.01 0.01 0.01                          !ROTOR TIP CLEARANCE
0.00001                                 !EQUIVALENT SAND ROUGHNESS
```

OUTPUT FILE FOR THE L.P. COMPRESSOR

From Compdes Program

Mid-Tandem Fan Engine

PROPERTY DISTRIBUTION THROUGH A 2 STAGE AXIAL COMPRESSOR WITH IGVs

CONSTANT OUTER RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)——————188.00
OVERALL PRESSURE RATIO———— 1.86
INLET AXIAL VELOCITY (m/s)——180.00
ISENTROPIC EFFICIENCY (%)———88.00
OVERALL TEMPERATURE RISE (K)— 63.48

INLET TOTAL TEPERATURE (K)---288.15
INLET TOTAL PRESSURE (atm)--- 1.00
ROTATIONAL SPEED (rpm)------- 5260.00
INLET IGV'S STA 1 STA 2 STA 3 STA 4 STA 5 STA 6 STA 7 STA 8 STA 9
TOTAL TEMPERATURE RISE (K)
30.00 33.48 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL TEMPERATURE (K)
288.15 288.15 318.15 351.63 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AVERAGE TOTAL TEMPERATURE (K)
303.15 334.89 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET STATIC TEMPERATURE (K)
272.00 269.86 295.61 331.78 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL PRESSURE (atm)
1.000 1.000 1.363 1.866 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET STATIC PRESSURE (atm)
0.817 0.795 1.054 1.522 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET DENSITY (kg/cubic m)
1.0610 1.0403 1.2589 1.6191 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET SPECIFIC HEAT (kJ/kg K)
1.0030 1.0030 1.0048 1.0075 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

AVERAGE SPECIFIC HEAT (kJ/kg K)
1.0038 1.0060 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET ANNULUS AREA (square m)
0.9844 1.0040 0.7467 0.5806 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET INNER RADIUS (cm)
32.32 31.34 42.44 48.27 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET OUTER RADIUS (cm)
64.64 64.64 64.64 64.64 0.00 0.00 0.00 0.00 0.00 0.00 0.00

BLADE HEIGHT (cm)
32.32 33.30 22.20 16.37 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET HUB/TIP RATIO
0.500 0.485 0.657 0.747 0.000 0.000 0.000 0.000 0.000 0.000 0.000

ABSOLUTE OUTLET VELOCITY (m/s)
180.00 191.55 212.84 200.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET SWIRL ANGLE (DEGREES)
0.00 20.00 20.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AXIAL VELOCITY (m/s)
180.00 180.00 200.00 200.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

PRESSURE RATIO
1.3633 1.3686 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

ENTHALPY RISE (kJ/kg)
30.114 33.681 0.000 0.000 0.000 0.000 0.000 0.000 0.000

COMPRESSOR POWER (kW)---------- 11993.516

OVERALL PRESSURE RATIO-------- 1.8659

L.P COMPRESSOR DESIGN !TITLE

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT OUTER RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9	
OUTSIDE RADIUS	64.64	64.64	64.64	0.00	0.00	0.00	0.00	0.00	0.00	0.00
HUB RADIUS	31.34	42.44	48.27	0.00	0.00	0.00	0.00	0.00	0.00	0.00
INLET HEIGHT	33.298	22.197	16.367	0.000	0.000	0.000	0.000	0.000	0.000	0.000
INLET HUB/TIP RATIO	0.4848	0.6566	0.7468	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	
ASPECT RATIO	3.0000	2.7500	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	
MID CHORD HEIGHT-ROTOR	31.074	21.092	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
MID CHORD HEIGHT-STATOR	25.379	18.121	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-ROTOR	10.358	7.670	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-STATOR	8.460	6.590	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL CHORD	8.757	6.420	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL SPACE	3.107	2.301	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL CHORD,	7.453	6.241	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL SPACE	2.538	1.977	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TOTAL STAGE LENGTH	21.856	16.939	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
NUMBER OF ROTOR BLADES	37.2	55.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	
NUMBER OF STATOR BLADES	55.1	75.7	0.0	0.0	0.0	0.0	0.0	0.0	0.0	

TOTAL COMPRESSOR LENGTH 38.795
STAGE 1

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	178.7	221.5	264.3	307.1	349.9	M/S
ABS. STATOR OUTLET VELOCITY (V0)	204.4	196.2	191.6	188.6	186.7	M/S
REL. ROTOR INLET VELOCITY (V1)	197.7	230.1	268.2	308.7	350.2	M/S
REL. ROTOR OUTLET VELOCITY (V2)	201.3	180.1	198.0	234.7	278.6	M/S
ABS. STATOR INLET VELOCITY (V3)	323.5	281.8	255.8	238.5	226.4	M/S
ABS. STATOR OUTLET VELOCITY (V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	180.0	180.0	180.0	180.0	180.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	180.0	180.0	180.0	180.0	180.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	0.782	0.721	0.647	0.568	0.491	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	0.732	0.860	0.937	0.987	1.021	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.60	0.70	0.81	0.94	1.06	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.99	0.84	0.76	0.70	0.66	
ROTOR DECELERATION (V2/V1)	1.02	0.78	0.74	0.76	0.80	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	-.04	0.39	0.45	0.42	0.37	
STATOR DECELERATION (V0/V3)	0.62	0.71	0.78	0.84	0.88	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	0.96	0.63	0.44	0.33	0.25	
FLOW COEFFICENT (VA/U)	1.01	0.81	0.68	0.59	0.51	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	28.3	23.5	20.0	17.4	15.4	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	24.5	38.5	47.8	54.3	59.1	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	-26.6	1.5	24.6	39.9	49.7	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	56.2	50.3	45.3	41.0	37.3	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	24.4	22.0	20.0	18.3	16.9	DEGREES
ROTOR DEFLECTION	51.0	37.1	23.2	14.4	9.3	DEGREES
STATOR DEFLECTION	31.8	28.3	25.3	22.7	20.4	DEGREES
REACTION	-2.3	33.4	53.2	65.4	73.3	PERCENT
SPACE CHORD RATIO (ROTOR)	0.631	0.767	0.800	0.791	0.773	
SPACE CHORD RATIO (STATOR)	0.451	0.577	0.700	0.820	0.938	
DIFFUSION FACTOR (ROTOR)	0.256	0.449	0.435	0.368	0.302	
DIFFUSION FACTOR (STATOR)	0.502	0.432	0.377	0.333	0.299	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.399	1.713	1.709	1.593	1.484	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.978	1.793	1.678	1.601	1.547	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00000	0.00000	0.00456	0.01947	
SHOCK LOSS COEFFICIENT (STATOR)	0.00635	0.00142	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.19E+06	.17E+06	.18E+06	.19E+06	.20E+06	
REYNLDS NUMBER (STATOR)	.17E+06	.17E+06	.17E+06	.17E+06	.18E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	1.012	1.030	1.044	1.055	1.063	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.04284	0.02342	0.02164	0.02303	0.02705
PROFILE LOSS COEFFICIENT (STATOR)	0.03313	0.02842	0.02477	0.02138	0.01924
SECONDARY LOSS COEFFICIENT (ROTOR)			0.01374		
SECONDARY LOSS COEFFICIENT (STATOR)			0.01263		
ANNULUS LOSS COEFFICIENT (ROTOR)			0.00614		
ANNULUS LOSS COEFFICIENT (STATOR)			0.00589		
TOTAL LOSS COEFFICIENT (ROTOR)			0.05228		
TOTAL LOSS COEFFICIENT (STATOR)			0.04546		

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.005

ROTOR ISENTROPIC EFFICIENCY 94.36 PERCENT

STAGE ISENTROPIC EFFICIENCY 90.02 PERCENT

STAGE PRESSURE RATIO 1.37

STAGE 2

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	236.8	265.9	294.9	323.9	353.0	M/S
ABS. STATOR OUTLET VELOCITY (V0)	219.6	215.7	212.8	210.7	209.0	M/S
REL. ROTOR INLET VELOCITY (V1)	247.7	272.5	298.9	326.2	354.1	M/S
REL. ROTOR OUTLET VELOCITY (V2)	200.1	205.9	223.3	247.6	275.6	M/S
ABS. STATOR INLET VELOCITY (V3)	315.2	295.1	279.7	267.8	258.3	M/S
ABS. STATOR OUTLET VELOCITY (V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	0.968	0.905	0.838	0.769	0.701	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	1.100	1.180	1.239	1.285	1.321	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.72	0.79	0.87	0.95	1.03	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.90	0.84	0.79	0.75	0.72	
ROTOR DECELERATION (V2/V1)	0.81	0.76	0.75	0.76	0.78	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.35	0.43	0.44	0.42	0.39	
STATOR DECELERATION (V0/V3)	0.63	0.68	0.71	0.75	0.77	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	0.65	0.51	0.42	0.35	0.29	
FLOW COEFFICENT (VA/U)	0.84	0.75	0.68	0.62	0.57	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	24.4	22.0	20.0	18.3	16.9	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	36.2	42.8	48.0	52.2	55.6	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	-1.9	13.7	26.4	36.1	43.5	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	50.6	47.3	44.4	41.7	39.3	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION	38.1	29.1	21.6	16.1	12.1	DEGREES
STATOR DEFLECTION	50.6	47.3	44.4	41.7	39.3	DEGREES
REACTION	29.4	44.0	54.5	62.3	68.2	PERCENT
SPACE CHORD RATIO (ROTOR)	0.753	0.790	0.800	0.796	0.787	

SPACE CHORD RATIO (STATOR)	0.558	0.630	0.700	0.769	0.837
DIFFUSION FACTOR (ROTOR)	0.425	0.442	0.417	0.378	0.336
DIFFUSION FACTOR (STATOR)	0.501	0.468	0.439	0.414	0.392
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.670	1.716	1.679	1.610	1.540
EQUIVALENT DIFFUSION FACTOR (STATOR)	2.028	1.935	1.865	1.811	1.769
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00000	0.00225	0.00488	0.02095
SHOCK LOSS COEFFICIENT (STATOR)	0.00351	0.00130	0.00000	0.00000	0.00000
REYNLDS NUMBER (ROTOR)	.16E+06	.16E+06	.17E+06	.17E+06	.18E+06
REYNLDS NUMBER (STATOR)	.17E+06	.18E+06	.18E+06	.18E+06	.19E+06
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000
AXIAL VELOCITY RATIO (STATOR)	1.000	1.000	1.000	1.000	1.000

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.02548	0.02231	0.02178	0.02307	0.02521
PROFILE LOSS COEFFICIENT (STATOR)	0.02800	0.02573	0.02373	0.02209	0.02074
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01247				
SECONDARY LOSS COEFFICIENT (STATOR)	0.02743				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00687				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00513				
TOTAL LOSS COEFFICIENT (ROTOR)	0.04854				
TOTAL LOSS COEFFICIENT (STATOR)	0.05758				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.003

ROTOR ISENTROPIC EFFICIENCY 94.33 PERCENT

STAGE ISENTROPIC EFFICIENCY 88.50 PERCENT

STAGE PRESSURE RATIO 1.37

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OVERALL PRESSURE RATIO 1.87

COMPRESSOR ISENTROPIC EFFICIENCY 88.67 PERCENT

COMPRESSOR POLYTROPIC EFFICIENCY 89.62 PERCENT

OUTPUT FILE FOR THE FAN

From Compdes Program

Mid-Tandem Fan Engine

PROPERTY DISTRIBUTION THROUGH A 2 STAGE AXIAL COMPRESSOR WITH IGVs

CONSTANT MEAN RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)————266.00
OVERALL PRESSURE RATIO———— 1.86
INLET AXIAL VELOCITY (m/s)——180.00
ISENTROPIC EFFICIENCY (%)———88.00
OVERALL TEMPERATURE RISE (K)— 63.48
INLET TOTAL TEPERATURE (K)—288.15
INLET TOTAL PRESSURE (atm)— 1.00
ROTATIONAL SPEED (rpm)——— 5050.00
INLET IGV'S STA 1 STA 2 STA 3 STA 4 STA 5 STA 6 STA 7 STA 8 STA 9

TOTAL TEMPERATURE RISE (K)
30.00 33.48 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL TEMPERATURE (K)
288.15 288.15 318.15 351.63 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AVERAGE TOTAL TEMPERATURE (K)
303.15 334.89 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET STATIC TEMPERATURE (K)
270.84 270.84 300.87 334.40 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL PRESSURE (atm)
1.000 1.000 1.363 1.866 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET STATIC PRESSURE (atm)
0.805 0.805 1.121 1.564 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET DENSITY (kg/cubic m)
1.0498 1.0498 1.3157 1.6514 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET SPECIFIC HEAT (kJ/kg K)
1.0030 1.0030 1.0048 1.0075 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

AVERAGE SPECIFIC HEAT (kJ/kg K)

1.0038 1.0060 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET ANNULUS AREA (square m)
1.4077 1.4077 1.1232 0.8949 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET INNER RADIUS (cm)
42.95 42.95 46.65 49.61 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET OUTER RADIUS (cm)
79.53 79.53 75.84 72.87 0.00 0.00 0.00 0.00 0.00 0.00 0.00

BLADE HEIGHT (cm)
36.59 36.59 29.19 23.26 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET HUB/TIP RATIO
0.540 0.540 0.615 0.681 0.000 0.000 0.000 0.000 0.000 0.000 0.000

ABSOLUTE OUTLET VELOCITY (m/s)
186.35 186.35 186.35 186.35 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET SWIRL ANGLE (DEGREES)
15.00 15.00 15.00 15.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AXIAL VELOCITY (m/s)
180.00 180.00 180.00 180.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

PRESSURE RATIO
1.3633 1.3686 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

ENTHALPY RISE (kJ/kg)
30.114 33.681 0.000 0.000 0.000 0.000 0.000 0.000 0.000

COMPRESSOR POWER (kW)---------- 16969.549

OVERALL PRESSURE RATIO--------- 1.8659

FAN COMPRESSOR DESIGN ITITLE

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT MEAN RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9
OUTSIDE RADIUS	79.53	75.84	72.87	0.00	0.00	0.00	0.00	0.00	0.00	0.00
HUB RADIUS	42.95	46.65	49.61	0.00	0.00	0.00	0.00	0.00	0.00	0.00
INLET HEIGHT	36.585	29.190	23.256	0.000	0.000	0.000	0.000	0.000	0.000	0.000
INLET HUB/TIP RATIO	0.5400	0.6151	0.6808	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
ASPECT RATIO	3.0000	2.7500	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000
MID CHORD HEIGHT-ROTOR	35.394	28.213	0.000	0.000	0.000	0.000	0.000	0.000	0.000
MID CHORD HEIGHT-STATOR	31.640	25.200	0.000	0.000	0.000	0.000	0.000	0.000	0.000
TRUE CHORD-ROTOR	11.798	10.259	0.000	0.000	0.000	0.000	0.000	0.000	0.000
TRUE CHORD-STATOR	10.547	9.164	0.000	0.000	0.000	0.000	0.000	0.000	0.000
ROTOR AXIAL CHORD	7.867	7.003	0.000	0.000	0.000	0.000	0.000	0.000	0.000
ROTOR AXIAL SPACE	3.539	3.078	0.000	0.000	0.000	0.000	0.000	0.000	0.000
STATOR AXIAL CHORD,	9.854	8.423	0.000	0.000	0.000	0.000	0.000	0.000	0.000
STATOR AXIAL SPACE	3.164	2.749	0.000	0.000	0.000	0.000	0.000	0.000	0.000
TOTAL STAGE LENGTH	24.424	21.252	0.000	0.000	0.000	0.000	0.000	0.000	0.000
NUMBER OF ROTOR BLADES	40.8	46.9	0.0	0.0	0.0	0.0	0.0	0.0	0.0
NUMBER OF STATOR BLADES	52.1	60.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0

TOTAL COMPRESSOR LENGTH 45.676

STAGE 1

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	230.3	277.1	323.9	370.7	417.4	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	192.4	188.6	186.3	184.9	183.8	M/S
REL. ROTOR INLET VELOCITY	(V1)	242.5	284.8	329.2	374.6	420.5	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	182.3	210.8	255.1	304.5	355.4	M/S
ABS. STATOR INLET VELOCITY	(V3)	270.0	245.7	230.0	219.2	211.5	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	186.3	186.3	186.3	186.3	186.3	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	180.0	180.0	180.0	180.0	180.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	180.0	180.0	180.0	180.0	180.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	0.697	0.614	0.530	0.448	0.374	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	0.888	0.959	1.003	1.032	1.053	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.74	0.86	1.00	1.13	1.27	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.80	0.72	0.67	0.64	0.61	
ROTOR DECELERATION	(V2/V1)	0.75	0.74	0.77	0.81	0.85	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	0.43	0.45	0.40	0.34	0.29	
STATOR DECELERATION	(V0/V3)	0.69	0.76	0.81	0.85	0.88	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	0.58	0.40	0.29	0.22	0.18	
FLOW COEFFICENT	(VA/U)	0.78	0.65	0.56	0.49	0.43	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	20.6	17.4	15.0	13.2	11.7	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	42.1	50.8	56.9	61.3	64.7	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	9.2	31.4	45.1	53.8	59.6	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	48.2	42.9	38.5	34.8	31.7	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	19.2	16.8	15.0	13.5	12.3	DEGREES
ROTOR DEFLECTION		32.9	19.4	11.7	7.5	5.1	DEGREES
STATOR DEFLECTION		29.0	26.1	23.5	21.3	19.4	DEGREES
REACTION		41.6	59.6	70.5	77.4	82.2	PERCENT

SPACE CHORD RATIO (ROTOR)	0.797	0.814	0.800	0.781	0.765	
SPACE CHORD RATIO (STATOR)	0.480	0.591	0.700	0.806	0.911	
DIFFUSION FACTOR (ROTOR)	0.467	0.418	0.340	0.274	0.222	
DIFFUSION FACTOR (STATOR)	0.428	0.375	0.334	0.302	0.277	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.754	1.678	1.545	1.440	1.367	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.729	1.622	1.552	1.504	1.470	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00213	0.00662	0.03165	0.06105	
SHOCK LOSS COEFFICIENT (STATOR)	0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.16E+06	.17E+06	.18E+06	.19E+06	.20E+06	
REYNLDS NUMBER (STATOR)	.27E+06	.28E+06	.28E+06	.28E+06	.29E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	0.978	0.991	1.000	1.007	1.012	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.02300	0.02220	0.02602	0.03061	0.03578
PROFILE LOSS COEFFICIENT (STATOR)	0.03303	0.02743	0.02420	0.02127	0.01927
SECONDARY LOSS COEFFICIENT (ROTOR)	0.00773				
SECONDARY LOSS COEFFICIENT (STATOR)	0.00972				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00839				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00593				
TOTAL LOSS COEFFICIENT (ROTOR)	0.06393				
TOTAL LOSS COEFFICIENT (STATOR)	0.04069				

EFFICIENCY LOSS DUE TO TIP CLEARANCE	0.024	
ROTOR ISENTROPIC EFFICIENCY	90.78	PERCENT
STAGE ISENTROPIC EFFICIENCY	87.56	PERCENT
STAGE PRESSURE RATIO	1.36	

STAGE 2

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	249.3	286.6	323.9	361.2	398.5	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	190.6	188.1	186.3	185.1	184.2	M/S
REL. ROTOR INLET VELOCITY	(V1)	259.3	293.7	329.2	365.3	401.8	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	184.7	208.7	243.4	282.4	323.1	M/S
ABS. STATOR INLET VELOCITY	(V3)	275.0	255.2	240.9	230.2	222.1	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	186.3	186.3	186.3	186.3	186.3	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	180.0	180.0	180.0	180.0	180.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	180.0	180.0	180.0	180.0	180.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	0.937	0.850	0.762	0.675	0.593	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	1.251	1.326	1.378	1.415	1.443	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.75	0.85	0.95	1.05	1.16	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.77	0.71	0.67	0.64	0.61	
ROTOR DECELERATION	(V2/V1)	0.71	0.71	0.74	0.77	0.80	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	0.49	0.49	0.45	0.40	0.35	
STATOR DECELERATION	(V0/V3)	0.68	0.73	0.77	0.81	0.84	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	0.58	0.44	0.35	0.28	0.23	
FLOW COEFFICENT	(VA/U)	0.72	0.63	0.56	0.50	0.45	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	19.2	16.8	15.0	13.5	12.3	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	46.0	52.2	56.9	60.5	63.4	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	12.9	30.4	42.3	50.4	56.1	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	49.1	45.1	41.6	38.6	35.9	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	15.0	15.0	15.0	15.0	15.0	DEGREES
ROTOR DEFLECTION		33.1	21.8	14.6	10.1	7.2	DEGREES
STATOR DEFLECTION		34.1	30.1	26.6	23.6	20.9	DEGREES
REACTION		45.7	58.9	67.8	74.1	78.8	PERCENT
SPACE CHORD RATIO (ROTOR)		0.812	0.814	0.800	0.783	0.767	
SPACE CHORD RATIO (STATOR)		0.527	0.614	0.700	0.784	0.868	
DIFFUSION FACTOR (ROTOR)		0.515	0.465	0.397	0.334	0.283	
DIFFUSION FACTOR (STATOR)		0.462	0.422	0.389	0.361	0.339	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.843	1.760	1.638	1.532	1.452	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.834	1.722	1.639	1.575	1.523	
SHOCK LOSS COEFFICIENT (ROTOR)		0.00000	0.00151	0.00492	0.02038	0.03284	
SHOCK LOSS COEFFICIENT (STATOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)		.20E+06	.21E+06	.22E+06	.23E+06	.24E+06	
REYNLDS NUMBER (STATOR)		.32E+06	.33E+06	.33E+06	.34E+06	.34E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.02010	0.02018	0.02265	0.02670	0.03121
PROFILE LOSS COEFFICIENT (STATOR)	0.02872	0.02516	0.02266	0.02089	0.01944
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01049				
SECONDARY LOSS COEFFICIENT (STATOR)	0.01255				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00854				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00626				
TOTAL LOSS COEFFICIENT (ROTOR)	0.05513				
TOTAL LOSS COEFFICIENT (STATOR)	0.04219				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.024

ROTOR ISENTROPIC EFFICIENCY 92.69 PERCENT

STAGE ISENTROPIC EFFICIENCY 89.40 PERCENT

STAGE PRESSURE RATIO 1.37

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OVERALL PRESSURE RATIO 1.86

COMPRESSOR ISENTROPIC EFFICIENCY 87.87 PERCENT

COMPRESSOR POLYTROPIC EFFICIENCY 88.88 PERCENT

OUTPUT FILE FOR THE LP. COMPRESSOR

From Compdes Program

Mid-Tandem Fan Engine

PROPERTY DISTRIBUTION THROUGH A 4 STAGE AXIAL COMPRESSOR WITH IGVs

CONSTANT MEAN RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)------------188.11
OVERALL PRESSURE RATIO--------- 1.80
INLET AXIAL VELOCITY (m/s)-----170.00
ISENTROPIC EFFICIENCY (%)-------88.00
OVERALL TEMPERATURE RISE (K)--- 73.20
INLET TOTAL TEPERATURE (K)---351.50
INLET TOTAL PRESSURE (atm)--- 1.86
ROTATIONAL SPEED (rpm)------ 5550.00
INLET IGV'S STA 1 STA 2 STA 3 STA 4 STA 5 STA 6 STA 7 STA 8 STA 9
TOTAL TEMPERATURE RISE (K)
18.30 18.30 18.30 18.30 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL TEMPERATURE (K)
351.50 351.50 369.80 388.10 406.40 424.70 0.00 0.00 0.00 0.00 0.00

AVERAGE TOTAL TEMPERATURE (K)
360.65 378.95 397.25 415.55 0.00 0.00 0.00 0.00 0.00

OUTLET STATIC TEMPERATURE (K)
337.16 337.16 355.48 373.81 392.14 410.48 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL PRESSURE (atm)
1.860 1.860 2.181 2.538 2.933 3.370 0.000 0.000 0.000 0.000 0.000

OUTLET STATIC PRESSURE (atm)
1.607 1.607 1.898 2.223 2.586 2.988 0.000 0.000 0.000 0.000 0.000

OUTLET DENSITY (kg/cubic m)
1.6827 1.6827 1.8849 2.1000 2.3281 2.5696 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET SPECIFIC HEAT (kJ/kg K)
1.0075 1.0075 1.0093 1.0114 1.0137 1.0162 0.0000 0.0000 0.0000 0.0000 0.0000
AVERAGE SPECIFIC HEAT (kJ/kg K)
1.0083 1.0103 1.0125 1.0149 0.0000 0.0000 0.0000 0.0000 0.0000
OUTLET ANNULUS AREA (square m)
0.6576 0.6576 0.5870 0.5269 0.4753 0.4306 0.0000 0.0000 0.0000 0.0000 0.0000
OUTLET INNER RADIUS (cm)
17.09 17.09 18.80 20.25 21.49 22.57 0.00 0.00 0.00 0.00 0.00
OUTLET OUTER RADIUS (cm)
48.84 48.84 47.14 45.69 44.44 43.36 0.00 0.00 0.00 0.00 0.00
BLADE HEIGHT (cm)
31.75 31.75 28.34 25.44 22.95 20.79 0.00 0.00 0.00 0.00 0.00
OUTLET HUB/TIP RATIO
0.350 0.350 0.399 0.443 0.484 0.521 0.000 0.000 0.000 0.000 0.000
ABSOLUTE OUTLET VELOCITY (m/s)
170.00 170.00 170.00 170.00 170.00 170.00 0.00 0.00 0.00 0.00 0.00

OUTLET SWIRL ANGLE (DEGREES)
0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AXIAL VELOCITY (m/s)
170.00 170.00 170.00 170.00 170.00 170.00 0.00 0.00 0.00 0.00 0.00

PRESSURE RATIO
1.1724 1.1637 1.1560 1.1490 0.0000 0.0000 0.0000 0.0000 0.0000

ENTHALPY RISE (kJ/kg)
18.453 18.488 18.529 18.573 0.000 0.000 0.000 0.000 0.000

COMPRESSOR POWER (kW)---------- 13928.124

OVERALL PRESSURE RATIO-------- 1.8121
LP COMPRESSOR DESIGN !TITLE

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT MEAN RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9	
OUTSIDE RADIUS	48.84	47.14	45.69	44.44	43.36	0.00	0.00	0.00	0.00	0.00
HUB RADIUS	17.09	18.80	20.25	21.49	22.57	0.00	0.00	0.00	0.00	0.00
INLET HEIGHT	31.746	28.340	25.438	22.946	20.789	0.000	0.000	0.000	0.000	0.000
INLET HUB/TIP RATIO	0.3500	0.3988	0.4432	0.4837	0.5206	0.0000	0.0000	0.0000	0.0000	
ASPECT RATIO	3.5000	3.2000	3.0000	2.8000	0.0000	0.0000	0.0000	0.0000	0.0000	
MID CHORD HEIGHT-ROTOR	31.148	27.829	24.998	22.564	0.000	0.000	0.000	0.000	0.000	
MID CHORD HEIGHT-STATOR	29.433	26.368	23.744	21.479	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-ROTOR	8.899	8.697	8.333	8.059	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-STATOR	8.409	8.240	7.915	7.671	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL CHORD	7.332	7.197	6.929	6.725	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL SPACE	2.670	2.609	2.500	2.418	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL CHORD,	8.329	8.150	7.816	7.567	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL SPACE	2.523	2.472	2.374	2.301	0.000	0.000	0.000	0.000	0.000	
TOTAL STAGE LENGTH	20.853	20.428	19.619	19.010	0.000	0.000	0.000	0.000	0.000	
NUMBER OF ROTOR BLADES	29.1	29.8	31.1	32.1	0.0	0.0	0.0	0.0	0.0	
NUMBER OF STATOR BLADES	35.2	35.9	37.4	38.6	0.0	0.0	0.0	0.0	0.0	

TOTAL COMPRESSOR LENGTH 79.911

STAGE 1

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	101.1	146.3	191.6	236.9	282.1	M/S
ABS. STATOR OUTLET VELOCITY (V0)	170.0	170.0	170.0	170.0	170.0	M/S
REL. ROTOR INLET VELOCITY (V1)	197.8	224.3	256.1	291.6	329.4	M/S
REL. ROTOR OUTLET VELOCITY (V2)	190.1	170.9	193.9	231.7	274.4	M/S
ABS. STATOR INLET VELOCITY (V3)	252.2	213.2	196.4	187.7	182.6	M/S
ABS. STATOR OUTLET VELOCITY (V4)	170.0	170.0	170.0	170.0	170.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	170.0	170.0	170.0	170.0	170.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	170.0	170.0	170.0	170.0	170.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	1.528	1.447	1.346	1.230	1.106	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	1.594	1.748	1.810	1.840	1.857	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.54	0.61	0.70	0.79	0.90	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.68	0.57	0.52	0.50	0.49	
ROTOR DECELERATION (V2/V1)	0.96	0.76	0.76	0.79	0.83	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.08	0.42	0.43	0.37	0.31	
STATOR DECELERATION (V0/V3)	0.67	0.80	0.87	0.91	0.93	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	1.84	0.88	0.51	0.34	0.24	
FLOW COEFFICENT (VA/U)	1.68	1.16	0.89	0.72	0.60	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	0.0	0.0	0.0	0.0	0.0	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	30.7	40.7	48.4	54.3	58.9	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	-26.6	5.9	28.8	42.8	51.7	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	47.6	37.1	30.0	25.1	21.4	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION	57.3	34.8	19.7	11.5	7.2	DEGREES
STATOR DEFLECTION	47.6	37.1	30.0	25.1	21.4	DEGREES
REACTION	7.9	56.0	74.4	83.2	88.2	PERCENT
SPACE CHORD RATIO (ROTOR)	0.510	0.704	0.800	0.845	0.866	
SPACE CHORD RATIO (STATOR)	0.364	0.535	0.700	0.861	1.021	
DIFFUSION FACTOR (ROTOR)	0.279	0.440	0.396	0.321	0.255	
DIFFUSION FACTOR (STATOR)	0.460	0.364	0.309	0.277	0.256	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.427	1.715	1.643	1.513	1.411	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.825	1.602	1.507	1.459	1.431	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00000	0.00000	0.00000	0.00320	
SHOCK LOSS COEFFICIENT (STATOR)	0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.32E+06	.31E+06	.32E+06	.34E+06	.36E+06	
REYNLDS NUMBER (STATOR)	.37E+06	.40E+06	.41E+06	.42E+06	.43E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.04348	0.02106	0.01931	0.02088	0.02407
PROFILE LOSS COEFFICIENT (STATOR)	0.03548	0.02783	0.02294	0.01909	0.01645

SECONDARY LOSS COEFFICIENT (ROTOR) 0.01106
SECONDARY LOSS COEFFICIENT (STATOR) 0.01314
ANNULUS LOSS COEFFICIENT (ROTOR) 0.00559
ANNULUS LOSS COEFFICIENT (STATOR) 0.00483
TOTAL LOSS COEFFICIENT (ROTOR) 0.04305
TOTAL LOSS COEFFICIENT (STATOR) 0.04233

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.029
ROTOR ISENTROPIC EFFICIENCY 93.13 PERCENT
STAGE ISENTROPIC EFFICIENCY 88.96 PERCENT
STAGE PRESSURE RATIO 1.17

STAGE 2

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	110.7	151.2	191.6	232.0	272.5	M/S
ABS. STATOR OUTLET VELOCITY (V0)	170.0	170.0	170.0	170.0	170.0	M/S
REL. ROTOR INLET VELOCITY (V1)	202.9	227.5	256.1	287.7	321.2	M/S
REL. ROTOR OUTLET VELOCITY (V2)	182.0	171.5	192.4	225.5	263.3	M/S
ABS. STATOR INLET VELOCITY (V3)	244.5	213.2	198.0	189.6	184.4	M/S
ABS. STATOR OUTLET VELOCITY (V4)	170.0	170.0	170.0	170.0	170.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	170.0	170.0	170.0	170.0	170.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	170.0	170.0	170.0	170.0	170.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	1.792	1.707	1.603	1.486	1.360	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	1.920	2.057	2.119	2.152	2.172	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.54	0.60	0.68	0.76	0.85	

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ABS. STATOR INLET MACH NUMBER (M3ABS)   0.65   0.56   0.52   0.49   0.48
ROTOR DECELERATION        (V2/V1)   0.90   0.75   0.75   0.78   0.82   ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT      (DP/D)   0.20   0.43   0.44   0.39   0.33
STATOR DECELERATION       (V0/V3)   0.70   0.80   0.86   0.90   0.92   STATOR DE HALLER NUMBER
STAGE LOADING   (DELTA H/U SQUARED)   1.59   0.85   0.53   0.36   0.26
FLOW COEFFICENT           (VA/U)   1.54   1.12   0.89   0.73   0.62
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)   0.0   0.0   0.0   0.0   0.0   DEGREES
REL. ROTOR AIR INLET ANGLE  (ALPHA1)  33.1   41.6   48.4   53.8   58.0   DEGREES
REL. ROTOR AIR OUTLET ANGLE  (ALPHA2)  -20.9   7.5   27.9   41.1   49.8   DEGREES
ABS. STATOR AIR INLET ANGLE  (ALPHA3)  46.0   37.1   30.9   26.3   22.8   DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)   0.0   0.0   0.0   0.0   0.0   DEGREES
ROTOR DEFLECTION                54.0   34.1   20.5   12.7   8.3   DEGREES
STATOR DEFLECTION               46.0   37.1   30.9   26.3   22.8   DEGREES
REACTION                20.6   57.4   73.5   81.9   86.9   PERCENT
SPACE CHORD RATIO (ROTOR)          0.559   0.717   0.800   0.841   0.862
SPACE CHORD RATIO (STATOR)         0.399   0.552   0.700   0.845   0.989
DIFFUSION FACTOR (ROTOR)          0.345   0.449   0.408   0.339   0.276
DIFFUSION FACTOR (STATOR)         0.448   0.369   0.321   0.290   0.270
EQUIVALENT DIFFUSION FACTOR (ROTOR)    1.524   1.731   1.662   1.541   1.442
EQUIVALENT DIFFUSION FACTOR (STATOR)   1.786   1.608   1.524   1.477   1.449
SHOCK LOSS COEFFICIENT (ROTOR)      0.00000 0.00000 0.00000 0.00000 0.00169
SHOCK LOSS COEFFICIENT (STATOR)     0.00000 0.00000 0.00000 0.00000 0.00000
REYNLDS NUMBER (ROTOR)          .36E+06  .35E+06  .37E+06  .39E+06  .41E+06
REYNLDS NUMBER (STATOR)         .43E+06  .46E+06  .48E+06  .49E+06  .49E+06
AXIAL VELOCITY RATIO (ROTOR)       1.000   1.000   1.000   1.000   1.000
AXIAL VELOCITY RATIO (STATOR)      1.000   1.000   1.000   1.000   1.000
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CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

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PROFILE LOSS COEFFICIENT (ROTOR)      0.03440  0.02011  0.01834  0.01997  0.02272
PROFILE LOSS COEFFICIENT (STATOR)     0.03263  0.02650  0.02216  0.01898  0.01658
SECONDARY LOSS COEFFICIENT (ROTOR)            0.01176
SECONDARY LOSS COEFFICIENT (STATOR)           0.01384
ANNULUS LOSS COEFFICIENT (ROTOR)             0.00603
ANNULUS LOSS COEFFICIENT (STATOR)            0.00524
TOTAL LOSS COEFFICIENT (ROTOR)              0.04123
TOTAL LOSS COEFFICIENT (STATOR)             0.04244

EFFICIENCY LOSS DUE TO TIP CLEARANCE           0.029
ROTOR ISENTROPIC EFFICIENCY               93.40   PERCENT
STAGE ISENTROPIC EFFICIENCY              89.14   PERCENT
STAGE PRESSURE RATIO                   1.16
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STAGE 3

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                      BLADE   25%   BLADE   75%   BLADE
                      ROOT          MID           TIP
                                    HEIGHT
BLADE SPEED              (U)  119.0   155.3   191.6   227.9   264.2   M/S
ABS. STATOR OUTLET VELOCITY   (V0)  170.0   170.0   170.0   170.0   170.0   M/S
REL. ROTOR INLET VELOCITY    (V1)  207.5   230.2   256.1   284.3   314.2   M/S
REL. ROTOR OUTLET VELOCITY   (V2)  177.3   171.9   190.7   220.0   253.5   M/S
ABS. STATOR INLET VELOCITY   (V3)  239.9   213.8   199.9   191.6   186.3   M/S
ABS. STATOR OUTLET VELOCITY   (V4)  170.0   170.0   170.0   170.0   170.0   M/S
AXIAL VELOCITY AT ROTOR INLET  (VA1)  170.0   170.0   170.0   170.0   170.0   M/S
AXIAL VELOCITY AT ROTOR OUTLET  (VA2)  170.0   170.0   170.0   170.0   170.0   M/S
STATIC PRESSURE AT ROTOR INLET  (PS1)   2.089   1.999   1.893   1.775   1.647   ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)   2.273   2.399   2.463   2.499   2.522   ATM
REL. ROTOR INLET MACH NUMBER  (M1REL)   0.54   0.59   0.66   0.73   0.81
ABS. STATOR INLET MACH NUMBER (M3ABS)   0.62   0.55   0.51   0.49   0.47
ROTOR DECELERATION        (V2/V1)   0.85   0.75   0.74   0.77   0.81   ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT      (DP/D)   0.27   0.44   0.45   0.40   0.35
STATOR DECELERATION       (V0/V3)   0.71   0.80   0.85   0.89   0.91   STATOR DE HALLER NUMBER
STAGE LOADING   (DELTA H/U SQUARED)   1.42   0.84   0.55   0.39   0.29
FLOW COEFFICENT           (VA/U)   1.43   1.09   0.89   0.75   0.64
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)   0.0   0.0   0.0   0.0   0.0   DEGREES
REL. ROTOR AIR INLET ANGLE  (ALPHA1)  35.0   42.4   48.4   53.3   57.2   DEGREES
REL. ROTOR AIR OUTLET ANGLE  (ALPHA2)  -16.5   8.6   27.0   39.4   47.9   DEGREES
ABS. STATOR AIR INLET ANGLE  (ALPHA3)  44.9   37.3   31.7   27.5   24.1   DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)   0.0   0.0   0.0   0.0   0.0   DEGREES
ROTOR DEFLECTION                51.5   33.9   21.5   13.9   9.4   DEGREES
STATOR DEFLECTION               44.9   37.3   31.7   27.5   24.1   DEGREES
REACTION                28.8   58.2   72.6   80.6   85.6   PERCENT
SPACE CHORD RATIO (ROTOR)          0.597   0.728   0.800   0.838   0.858
SPACE CHORD RATIO (STATOR)         0.428   0.566   0.700   0.831   0.961
DIFFUSION FACTOR (ROTOR)          0.389   0.458   0.419   0.357   0.297
DIFFUSION FACTOR (STATOR)         0.443   0.377   0.334   0.304   0.284
EQUIVALENT DIFFUSION FACTOR (ROTOR)    1.595   1.747   1.683   1.571   1.473
EQUIVALENT DIFFUSION FACTOR (STATOR)   1.765   1.618   1.541   1.496   1.467
SHOCK LOSS COEFFICIENT (ROTOR)      0.00000 0.00000 0.00000 0.00000 0.00000
SHOCK LOSS COEFFICIENT (STATOR)     0.00000 0.00000 0.00000 0.00000 0.00000
REYNLDS NUMBER (ROTOR)          .40E+06  .40E+06  .41E+06  .44E+06  .46E+06
REYNLDS NUMBER (STATOR)         .49E+06  .52E+06  .53E+06  .54E+06  .55E+06
AXIAL VELOCITY RATIO (ROTOR)       1.000   1.000   1.000   1.000   1.000
AXIAL VELOCITY RATIO (STATOR)      1.000   1.000   1.000   1.000   1.000
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CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENC

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PROFILE LOSS COEFFICIENT (ROTOR)      0.02925  0.01942  0.01789  0.01921  0.02158
PROFILE LOSS COEFFICIENT (STATOR)     0.03049  0.02543  0.02170  0.01889  0.01671
SECONDARY LOSS COEFFICIENT (ROTOR)            0.01253
SECONDARY LOSS COEFFICIENT (STATOR)           0.01459
ANNULUS LOSS COEFFICIENT (ROTOR)             0.00633
ANNULUS LOSS COEFFICIENT (STATOR)            0.00553
TOTAL LOSS COEFFICIENT (ROTOR)              0.04033
TOTAL LOSS COEFFICIENT (STATOR)             0.04277
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EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.029
ROTOR ISENTROPIC EFFICIENCY 93.53 PERCENT
STAGE ISENTROPIC EFFICIENCY 89.16 PERCENT
STAGE PRESSURE RATIO 1.16

STAGE 4

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	126.0	158.8	191.6	224.4	257.2	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	170.0	170.0	170.0	170.0	170.0	M/S
REL. ROTOR INLET VELOCITY	(V1)	211.6	232.6	256.1	281.5	308.3	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	174.1	172.4	189.6	215.5	245.4	M/S
ABS. STATOR INLET VELOCITY	(V3)	236.0	214.0	'201.2	193.3	188.0	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	170.0	170.0	170.0	170.0	170.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	170.0	170.0	170.0	170.0	170.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	170.0	170.0	170.0	170.0	170.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	2.419	2.326	2.218	2.099	1.970	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	2.664	2.782	2.846	2.884	2.910	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.53	0.59	0.65	0.71	0.78	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.59	0.53	0.50	0.48	0.47	
ROTOR DECELERATION	(V2/V1)	0.82	0.74	0.74	0.77	0.80	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	0.32	0.45	0.45	0.41	0.37	
STATOR DECELERATION	(V0/V3)	0.72	0.79	0.84	0.88	0.90	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	1.30	0.82	0.56	0.41	0.31	
FLOW COEFFICENT	(VA/U)	1.35	1.07	0.89	0.76	0.66	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	0.0	0.0	0.0	0.0	0.0	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	36.6	43.1	48.4	52.9	56.5	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	-12.5	9.6	26.3	37.9	46.1	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	43.9	37.4	32.4	28.4	25.3	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION		49.1	33.4	22.2	14.9	10.4	DEGREES
STATOR DEFLECTION		43.9	37.4	32.4	28.4	25.3	DEGREES
REACTION		35.0	59.1	71.9	79.5	84.4	PERCENT
SPACE CHORD RATIO (ROTOR)		0.627	0.737	0.800	0.835	0.854	
SPACE CHORD RATIO (STATOR)		0.454	0.579	0.700	0.819	0.937	
DIFFUSION FACTOR (ROTOR)		0.420	0.465	0.428	0.371	0.315	
DIFFUSION FACTOR (STATOR)		0.437	0.381	0.343	0.315	0.296	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.650	1.759	1.697	1.594	1.501	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.747	1.624	1.554	1.511	1.483	
SHOCK LOSS COEFFICIENT (ROTOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
SHOCK LOSS COEFFICIENT (STATOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)		.44E+06	.45E+06	.46E+06	.49E+06	.51E+06	
REYNLDS NUMBER (STATOR)		.56E+06	.58E+06	.60E+06	.61E+06	.62E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.02581	0.01882	0.01751	0.01859	0.02062
PROFILE LOSS COEFFICIENT (STATOR)	0.02878	0.02452	0.02128	0.01878	0.01680
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01311				
SECONDARY LOSS COEFFICIENT (STATOR)	0.01514				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00671				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00588				
TOTAL LOSS COEFFICIENT (ROTOR)	0.04008				
TOTAL LOSS COEFFICIENT (STATOR)	0.04306				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.028
ROTOR ISENTROPIC EFFICIENCY 93.55 PERCENT
STAGE ISENTROPIC EFFICIENCY 89.10 PERCENT
STAGE PRESSURE RATIO 1.15
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OVERALL PRESSURE RATIO 1.81
COMPRESSOR ISENTROPIC EFFICIENCY 88.06 PERCENT
COMPRESSOR POLYTROPIC EFFICIENCY 89.01 PERCENT

OUTPUT FILE FOR THE H.P. COMPRESSOR

From Compdes Program

Mid-Tandem Fan Engine

PROPERTY DISTRIBUTION THROUGH A 3 STAGE AXIAL COMPRESSOR WITHOUT IGVs

CONSTANT OUTER RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)——————188.11
OVERALL PRESSURE RATIO———— 2.54
INLET AXIAL VELOCITY (m/s)——200.00
ISENTROPIC EFFICIENCY (%)———88.00
OVERALL TEMPERATURE RISE (K)—143.27
INLET TOTAL TEPERATURE (K)—424.15
INLET TOTAL PRESSURE (atm)— 3.35
ROTATIONAL SPEED (rpm)——— 8200.00

INLET IGV'S	STA 1	STA 2	STA 3	STA 4	STA 5	STA 6	STA 7	STA 8	STA 9
TOTAL TEMPERATURE RISE (K)	45.27	50.00	48.00	0.00	0.00	0.00	0.00	0.00	0.00

OUTLET TOTAL TEMPERATURE (K)
424.15 424.15 469.42 519.42 567.42 0.00 0.00 0.00 0.00 0.00 0.00

AVERAGE TOTAL TEMPERATURE (K)
446.79 494.42 543.42 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET STATIC TEMPERATURE (K)
404.47 403.05 448.47 499.45 548.24 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL PRESSURE (atm)
3.348 3.348 4.629 6.412 8.543 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET STATIC PRESSURE (atm)
2.830 2.795 3.933 5.568 7.540 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET DENSITY (kg/cubic m)
2.4699 2.4480 3.0964 3.9359 4.8554 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET SPECIFIC HEAT (kJ/kg K)
1.0161 1.0161 1.0233 1.0327 1.0428 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

AVERAGE SPECIFIC HEAT (kJ/kg K)
1.0196 1.0278 1.0376 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET ANNULUS AREA (square m)
0.3808 0.3842 0.3038 0.2390 0.1937 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET INNER RADIUS (cm)
34.13 33.97 37.55 40.20 41.95 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET OUTER RADIUS (cm)
48.75 48.75 48.75 48.75 48.75 0.00 0.00 0.00 0.00 0.00 0.00

BLADE HEIGHT (cm)
14.63 14.78 11.20 8.55 6.80 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET HUB/TIP RATIO
0.700 0.697 0.770 0.825 0.861 0.000 0.000 0.000 0.000 0.000 0.000

ABSOLUTE OUTLET VELOCITY (m/s)
200.00 207.06 207.06 203.09 200.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET SWIRL ANGLE (DEGREES)
0.00 15.00 15.00 10.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AXIAL VELOCITY (m/s)
200.00 200.00 200.00 200.00 200.00 0.00 0.00 0.00 0.00 0.00 0.00

PRESSURE RATIO
1.3825 1.3852 1.3325 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

ENTHALPY RISE (kJ/kg)
46.156 51.392 49.806 0.000 0.000 0.000 0.000 0.000 0.000

COMPRESSOR POWER (kW)---------- 27718.701

OVERALL PRESSURE RATIO-------- 2.5517

H.P COMPRESSOR DESIGN ITITLE

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT OUTER RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9	
OUTSIDE RADIUS	48.75	48.75	48.75	48.75	0.00	0.00	0.00	0.00	0.00	0.00
HUB RADIUS	33.97	37.55	40.20	41.95	0.00	0.00	0.00	0.00	0.00	0.00
INLET HEIGHT	14.785	11.204	8.551	6.798	0.000	0.000	0.000	0.000	0.000	0.000
INLET HUB/TIP RATIO	0.6967	0.7702	0.8246	0.8606	0.0000	0.0000	0.0000	0.0000	0.0000	
ASPECT RATIO	2.0000	2.0000	2.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	
MID CHORD HEIGHT-ROTOR	14.177	10.761	8.282	0.000	0.000	0.000	0.000	0.000	0.000	
MID CHORD HEIGHT-STATOR	12.353	9.410	7.392	0.000	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-ROTOR	7.089	5.381	4.141	0.000	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-STATOR	6.177	4.705	3.696	0.000	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL CHORD	4.939	3.674	2.614	0.000	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL SPACE	2.127	1.614	1.242	0.000	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL CHORD,	5.641	4.311	3.547	0.000	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL SPACE	1.853	1.412	1.109	0.000	0.000	0.000	0.000	0.000	0.000	
TOTAL STAGE LENGTH	14.559	11.011	8.512	0.000	0.000	0.000	0.000	0.000	0.000	
NUMBER OF ROTOR BLADES	46.2	63.3	84.6	0.0	0.0	0.0	0.0	0.0	0.0	
NUMBER OF STATOR BLADES	57.7	78.4	102.1	0.0	0.0	0.0	0.0	0.0	0.0	
TOTAL COMPRESSOR LENGTH	34.082									

STAGE 1

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	294.3	324.7	355.2	385.6	416.0	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	210.2	208.4	207.1	206.0	205.2	M/S
REL. ROTOR INLET VELOCITY	(V1)	304.5	332.9	361.9	391.2	420.8	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	212.3	234.6	262.7	293.9	326.6	M/S
ABS. STATOR INLET VELOCITY	(V3)	299.6	284.4	272.3	262.7	254.8	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	200.0	200.0	200.0	200.0	200.0	M/S

AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	2.289	2.136	1.980	1.826	1.675	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	3.281	3.402	3.496	3.569	3.628	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.76	0.83	0.90	0.97	1.05	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.73	0.69	0.66	0.63	0.61	
ROTOR DECELERATION (V2/V1)	0.70	0.70	0.73	0.75	0.78	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.51	0.50	0.47	0.44	-0.40	
STATOR DECELERATION (V0/V3)	0.67	0.70	0.73	0.76	0.79	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	0.54	0.44	0.37	0.31	0.27	
FLOW COEFFICENT (VA/U)	0.68	0.62	0.56	0.52	0.48	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	17.9	16.3	15.0	13.9	12.9	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	48.9	53.1	56.4	59.3	61.6	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	19.6	31.5	40.4	47.1	52.2	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	48.1	45.3	42.7	40.4	38.3	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	17.0	15.9	15.0	14.2	13.4	DEGREES
ROTOR DEFLECTION	29.4	21.6	16.0	12.1	9.4	DEGREES
STATOR DEFLECTION	31.1	29.4	27.7	26.3	24.9	DEGREES
REACTION	51.1	59.8	66.4	71.5	75.5	PERCENT
SPACE CHORD RATIO (ROTOR)	0.815	0.811	0.800	0.787	0.774	
SPACE CHORD RATIO (STATOR)	0.609	0.680	0.750	0.819	0.887	
DIFFUSION FACTOR (ROTOR)	0.515	0.470	0.419	0.370	0.327	
DIFFUSION FACTOR (STATOR)	0.494	0.468	0.446	0.427	0.410	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.850	1.772	1.678	1.592	1.520	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.796	1.735	1.686	1.647	1.615	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00098	0.00339	0.00583	0.01686	
SHOCK LOSS COEFFICIENT (STATOR)	0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.32E+06	.33E+06	.34E+06	.36E+06	.37E+06	
REYNLDS NUMBER (STATOR)	.43E+06	.43E+06	.44E+06	.44E+06	.45E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	0.956	0.962	0.966	0.970	0.973	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.01731	0.01792	0.01981	0.02183	0.02467
PROFILE LOSS COEFFICIENT (STATOR)	0.02203	0.02008	0.01900	0.01731	0.01636
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01172				
SECONDARY LOSS COEFFICIENT (STATOR)	0.01461				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.01130				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00851				
TOTAL LOSS COEFFICIENT (ROTOR)	0.04874				
TOTAL LOSS COEFFICIENT (STATOR)	0.04207				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.002

ROTOR ISENTROPIC EFFICIENCY 94.15 PERCENT

STAGE ISENTROPIC EFFICIENCY 91.09 PERCENT

STAGE PRESSURE RATIO 1.39

STAGE 2

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	324.3	347.4	370.5	393.6	416.7	M/S
ABS. STATOR OUTLET VELOCITY (V0)	209.2	208.0	207.1	206.3	205.6	M/S
REL. ROTOR INLET VELOCITY (V1)	330.5	352.5	374.8	397.2	419.8	M/S
REL. ROTOR OUTLET VELOCITY (V2)	222.0	241.1	263.1	286.9	311.9	M/S
ABS. STATOR INLET VELOCITY (V3)	303.3	292.1	282.6	274.4	267.4	M/S
ABS. STATOR OUTLET VELOCITY (V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	3.117	2.960	2.802	2.645	2.490	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	4.698	4.814	4.911	4.992	5.061	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.78	0.83	0.89	0.94	0.99	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.70	0.67	0.65	0.63	0.61	
ROTOR DECELERATION (V2/V1)	0.67	0.68	0.70	0.72	0.74	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.55	0.53	0.51	0.48	0.45	
STATOR DECELERATION (V0/V3)	0.66	0.68	0.71	0.73	0.75	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	0.51	0.45	0.39	0.35	0.31	
FLOW COEFFICENT (VA/U)	0.62	0.58	0.54	0.51	0.48	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	17.0	15.9	15.0	14.2	13.4	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	52.8	55.4	57.7	59.8	61.5	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	25.7	33.9	40.5	45.8	50.1	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	48.7	46.8	44.9	43.2	41.6	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	11.0	10.5	10.0	9.6	9.2	DEGREES
ROTOR DEFLECTION	27.0	21.5	17.2	14.0	11.4	DEGREES
STATOR DEFLECTION	37.7	36.3	34.9	33.6	32.4	DEGREES
REACTION	55.4	61.1	65.8	69.7	73.0	PERCENT
SPACE CHORD RATIO (ROTOR)	0.821	0.812	0.800	0.788	0.776	
SPACE CHORD RATIO (STATOR)	0.648	0.699	0.750	0.800	0.850	
DIFFUSION FACTOR (ROTOR)	0.535	0.495	0.454	0.414	0.377	
DIFFUSION FACTOR (STATOR)	0.519	0.502	0.486	0.472	0.458	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.895	1.819	1.740	1.666	1.602	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.909	1.861	1.820	1.786	1.756	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00113	0.00288	0.00466	0.00644	
SHOCK LOSS COEFFICIENT (STATOR)	0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.33E+06	.35E+06	.36E+06	.37E+06	.38E+06	
REYNLDS NUMBER (STATOR)	.45E+06	.46E+06	.46E+06	.46E+06	.47E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	0.982	0.983	0.985	0.986	0.987	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.01720	0.01800	0.01927	0.02078	0.02290
PROFILE LOSS COEFFICIENT (STATOR)	0.02040	0.01911	0.01806	0.01721	0.01649
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01379				
SECONDARY LOSS COEFFICIENT (STATOR)	0.02118				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.01117				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00781				
TOTAL LOSS COEFFICIENT (ROTOR)	0.04761				
TOTAL LOSS COEFFICIENT (STATOR)	0.04725				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.002

ROTOR ISENTROPIC EFFICIENCY 94.47 PERCENT

STAGE ISENTROPIC EFFICIENCY 91.14 PERCENT

STAGE PRESSURE RATIO 1.39

STAGE 3

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	346.4	364.1	381.9	399.7	417.5	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	203.7	203.4	203.1	202.8	202.6	M/S
REL. ROTOR INLET VELOCITY	(V1)	366.8	383.4	400.2	417.1	434.0	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	250.7	268.1	286.3	305.2	324.5	M/S
ABS. STATOR INLET VELOCITY	(V3)	279.5	272.9	267.1	261.9	257.3	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	4.173	4.016	3.859	3.703	3.548	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	6.758	6.839	6.909	6.971	7.025	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.82	0.86	0.90	0.94	0.97	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.61	0.59	0.58	0.57	0.56	
ROTOR DECELERATION	(V2/V1)	0.68	0.70	0.72	0.73	0.75	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	0.53	0.51	0.49	0.46	0.44	
STATOR DECELERATION	(V0/V3)	0.72	0.73	0.75	0.76	0.78	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	0.45	0.41	0.37	0.34	0.31	
FLOW COEFFICENT	(VA/U)	0.58	0.55	0.52	0.50	0.48	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	11.0	10.5	10.0	9.6	9.2	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	57.0	58.6	60.0	61.3	62.6	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	37.1	41.7	45.7	49.1	52.0	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	44.3	42.9	41.5	40.2	39.0	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION		19.9	16.8	14.3	12.3	10.6	DEGREES
STATOR DEFLECTION		44.3	42.9	41.5	40.2	39.0	DEGREES
REACTION		66.2	69.4	72.2	74.6	76.7	PERCENT
SPACE CHORD RATIO (ROTOR)		0.814	0.807	0.800	0.793	0.786	
SPACE CHORD RATIO (STATOR)		0.679	0.715	0.750	0.785	0.820	
DIFFUSION FACTOR (ROTOR)		0.490	0.457	0.426	0.397	0.370	
DIFFUSION FACTOR (STATOR)		0.474	0.462	0.450	0.439	0.429	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.807	1.745	1.686	1.633	1.586	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.854	1.825	1.799	1.776	1.756	
SHOCK LOSS COEFFICIENT (ROTOR)		0.00078	0.00203	0.00329	0.00456	0.00583	
SHOCK LOSS COEFFICIENT (STATOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)		.35E+06	.36E+06	.37E+06	.37E+06	.38E+06	
REYNLDS NUMBER (STATOR)		.52E+06	.52E+06	.53E+06	.53E+06	.53E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.01848	0.01950	0.02053	0.02207	0.02372
PROFILE LOSS COEFFICIENT (STATOR)	0.02028	0.01948	0.01879	0.01818	0.01764
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01231				
SECONDARY LOSS COEFFICIENT (STATOR)	0.02594				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.01234				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00733				
TOTAL LOSS COEFFICIENT (ROTOR)	0.04881				
TOTAL LOSS COEFFICIENT (STATOR)	0.05215				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.001

ROTOR ISENTROPIC EFFICIENCY 93.50 PERCENT

STAGE ISENTROPIC EFFICIENCY 90.05 PERCENT

STAGE PRESSURE RATIO 1.34

****

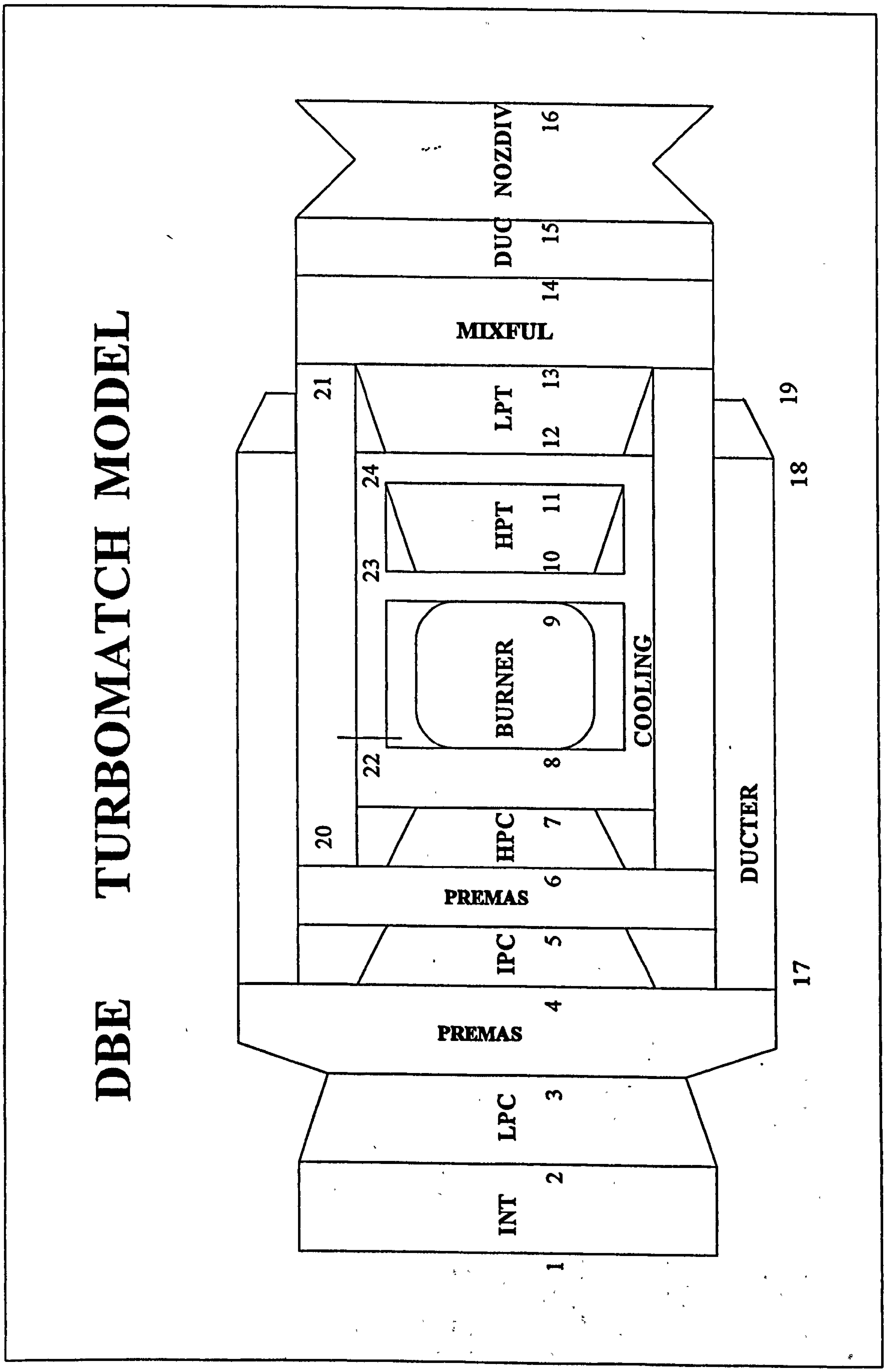
OVERALL PRESSURE RATIO 2.59

COMPRESSOR ISENTROPIC EFFICIENCY 89.01 PERCENT

COMPRESSOR POLYTROPIC EFFICIENCY 90.33 PERCENT

APPENDIX D

DOUBLE BYPASS ENGINE


DBE TURBOMATCH MODEL
INT
LPC
PREMAS
IPC
PREMAS
HPC
BURNER
COOLING
HPT
LPT
MIXFUL
DUC
NOZDIV
DUCTER
1
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TURBOMATCH INPUT FILE

Double Bypass Engine

```
LP MODE FOR THE DOUBLE BYPASS ENGINE////
OP SI KE VA FP
 2
-10 -5 0 5 10 15
 0.53 0.63 0.74 0.79 0.84 0.89 0.95 1.00 1.05 1.16
 0.95 120. 0.810 0.97 116. 0.110 1.01 104. 0.321 1.05  85. 0.911 1.08  64. 0.814
 1.00 126. 0.159 1.02 122. 0.469 1.05 112. 0.815 1.09  96. 0.924 1.11  77. 0.817
 1.04 132. 0.706 1.08 129. 0.845 1.11 120. 0.908 1.14 106. 0.909 1.15  90. 0.817
 1.07 136. 0.799 1.12 133. 0.887 1.14 125. 0.913 1.17 112. 0.897 1.18  98. 0.817
 1.09 140. 0.832 1.15 137. 0.898 1.17 130. 0.908 1.19 119. 0.885 1.20 106. 0.816
 1.11 144. 0.835 1.18 141. 0.894 1.21 135. 0.897 1.22 125. 0.871 1.23 114. 0.810
 1.13 148. 0.826 1.21 146. 0.884 1.24 140. 0.885 1.26 132. 0.859 1.26 122. 0.803
 1.15 152. 0.812 1.24 150. 0.870 1.27 145. 0.872 1.29 139. 0.848 1.30 131. 0.799
 1.18 155. 0.794 1.27 154. 0.851 1.31 151. 0.859 1.33 146. 0.839 1.33 139. 0.799
 1.23 163. 0.741 1.31 162. 0.796 1.39 160. 0.820 1.41 157. 0.812 1.42 154. 0.788
 0.93 118. 0.310 0.95 114. 0.430 1.00 101. 0.201 1.05  82. 0.911 1.07  59. 0.790
 0.98 124. 0.201 1.00 120. 0.048 1.04 108. 0.752 1.09  91. 0.927 1.11  71. 0.793
 1.02 129. 0.526 1.06 126. 0.775 1.09 116. 0.898 1.13 101. 0.914 1.15  83. 0.798
 1.05 133. 0.739 1.09 129. 0.860 1.12 120. 0.915 1.16 106. 0.905 1.17  90. 0.801
 1.07 136. 0.814 1.13 133. 0.894 1.16 125. 0.917 1.18 112. 0.894 1.20  97. 0.803
 1.10 140. 0.843 1.16 137. 0.904 1.19 129. 0.912 1.21 118. 0.883 1.22 104. 0.803
 1.12 143. 0.848 1.19 141. 0.902 1.22 134. 0.903 1.24 124. 0.873 1.25 112. 0.801
 1.14 147. 0.843 1.22 145. 0.894 1.26 139. 0.894 1.28 130. 0.863 1.29 119. 0.798
 1.17 151. 0.833 1.26 149. 0.884 1.29 144. 0.883 1.32 136. 0.853 1.32 127. 0.793
 1.23 158. 0.809 1.32 157. 0.856 1.38 154. 0.863 1.40 149. 0.843 1.41 144. 0.802
 0.91 117. 0.231 0.93 112. 0.261 0.98  99. 0.311 1.04  79. 0.907 1.07  55. 0.769
 0.96 122. 0.130 0.98 117. 0.260 1.03 105. 0.649 1.08  87. 0.928 1.10  66. 0.770
 1.01 127. 0.183 1.04 123. 0.651 1.08 112. 0.882 1.12  96. 0.917 1.14  77. 0.774
 1.03 129. 0.611 1.07 126. 0.808 1.11 116. 0.911 1.15 101. 0.907 1.16  83. 0.777
 1.05 132. 0.757 1.10 129. 0.873 1.14 120. 0.921 1.17 106. 0.897 1.19  89. 0.780
 1.08 136. 0.820 1.14 133. 0.900 1.17 124. 0.920 1.20 111. 0.888 1.22  95. 0.783
 1.10 139. 0.845 1.17 137. 0.908 1.20 128. 0.915 1.23 116. 0.878 1.24 102. 0.785
 1.12 143. 0.851 1.20 140. 0.907 1.24 133. 0.907 1.27 122. 0.870 1.28 109. 0.785
 1.15 147. 0.849 1.24 144. 0.902 1.28 138. 0.899 1.30 128. 0.862 1.31 116. 0.784
 1.20 154. 0.833 1.31 152. 0.885 1.36 148. 0.885 1.38 141. 0.855 1.39 133. 0.795
 0.91 117. 0.231 0.93 112. 0.261 0.98  99. 0.311 1.04  79. 0.907 1.07  55. 0.769
 0.96 122. 0.130 0.98 117. 0.260 1.03 105. 0.649 1.08  87. 0.928 1.10  66. 0.770
 1.01 127. 0.183 1.04 123. 0.651 1.08 112. 0.882 1.12  96. 0.917 1.14  77. 0.774
 1.03 129. 0.611 1.07 126. 0.808 1.11 116. 0.911 1.15 101. 0.907 1.16  83. 0.777
 1.05 132. 0.757 1.10 129. 0.873 1.14 120. 0.921 1.17 106. 0.897 1.19  89. 0.780
 1.08 136. 0.820 1.14 133. 0.900 1.17 124. 0.920 1.20 111. 0.888 1.22  95. 0.783
 1.10 139. 0.845 1.17 137. 0.908 1.20 128. 0.915 1.23 116. 0.878 1.24 102. 0.785
 1.12 143. 0.851 1.20 140. 0.907 1.24 133. 0.907 1.27 122. 0.870 1.28 109. 0.785
 1.15 147. 0.849 1.24 144. 0.902 1.28 138. 0.899 1.30 128. 0.862 1.31 116. 0.784
 1.20 154. 0.833 1.31 152. 0.885 1.36 148. 0.885 1.38 141. 0.855 1.39 133. 0.795
 0.91 117. 0.231 0.93 112. 0.261 0.98  99. 0.311 1.04  79. 0.907 1.07  55. 0.769
 0.96 122. 0.130 0.98 117. 0.260 1.03 105. 0.649 1.08  87. 0.928 1.10  66. 0.770
 1.01 127. 0.183 1.04 123. 0.651 1.08 112. 0.882 1.12  96. 0.917 1.14  77. 0.774
 1.03 129. 0.611 1.07 126. 0.808 1.11 116. 0.911 1.15 101. 0.907 1.16  83. 0.777
 1.05 132. 0.757 1.10 129. 0.873 1.14 120. 0.921 1.17 106. 0.897 1.19  89. 0.780
 1.08 136. 0.820 1.14 133. 0.900 1.17 124. 0.920 1.20 111. 0.888 1.22  95. 0.783
 1.10 139. 0.845 1.17 137. 0.908 1.20 128. 0.915 1.23 116. 0.878 1.24 102. 0.785
 1.12 143. 0.851 1.20 140. 0.907 1.24 133. 0.907 1.27 122. 0.870 1.28 109. 0.785
 1.15 147. 0.849 1.24 144. 0.902 1.28 138. 0.899 1.30 128. 0.862 1.31 116. 0.784
 1.20 154. 0.833 1.31 152. 0.885 1.36 148. 0.885 1.38 141. 0.855 1.39 133. 0.795
 0.91 117. 0.231 0.93 112. 0.261 0.98  99. 0.311 1.04  79. 0.907 1.07  55. 0.769
 0.96 122. 0.130 0.98 117. 0.260 1.03 105. 0.649 1.08  87. 0.928 1.10  66. 0.770
 1.01 127. 0.183 1.04 123. 0.651 1.08 112. 0.882 1.12  96. 0.917 1.14  77. 0.774
 1.03 129. 0.611 1.07 126. 0.808 1.11 116. 0.911 1.15 101. 0.907 1.16  83. 0.777
 1.05 132. 0.757 1.10 129. 0.873 1.14 120. 0.921 1.17 106. 0.897 1.19  89. 0.780
 1.08 136. 0.820 1.14 133. 0.900 1.17 124. 0.920 1.20 111. 0.888 1.22  95. 0.783
 1.10 139. 0.845 1.17 137. 0.908 1.20 128. 0.915 1.23 116. 0.878 1.24 102. 0.785
 1.12 143. 0.851 1.20 140. 0.907 1.24 133. 0.907 1.27 122. 0.870 1.28 109. 0.785
 1.15 147. 0.849 1.24 144. 0.902 1.28 138. 0.899 1.30 128. 0.862 1.31 116. 0.784
 1.20 154. 0.833 1.31 152. 0.885 1.36 148. 0.885 1.38 141. 0.855 1.39 133. 0.795
 3
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 2.78 102. 0.745 3.47 101. 0.891 3.67  98. 0.885 3.82  93. 0.864 3.86  88. 0.832
 2.92 109. 0.723 3.88 108. 0.888 4.17 105. 0.884 4.36 101. 0.864 4.42  96. 0.834
 3.62 115. 0.790 4.98 115. 0.856 5.26 115. 0.851 5.41 115. 0.845 5.50 114. 0.840
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 1.29 54. 0.462 1.59 53. 0.830 1.66 49. 0.835 1.73 44. 0.814 1.75 38. 0.767
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1.68 65. 0.665 1.97 64. 0.856 2.04 60. 0.849 2.10 55. 0.823 2.13 49. 0.781
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2.14 86. 0.657 2.77 85. 0.881 2.88 81. 0.871 2.95 76. 0.845 2.98 71. 0.809
1.35 94. 0.218 3.10 93. 0.887 3.25 89. 0.879 3.35 85. 0.855 3.38 79. 0.820
2.78 102. 0.745 3.47 101. 0.891 3.67  98. 0.885 3.82  93. 0.864 3.86  88. 0.832
2.92 109. 0.723 3.88 108. 0.888 4.17 105. 0.884 4.36 101. 0.864 4.42  96. 0.834
3.62 115. 0.790 4.98 115. 0.856 5.26 115. 0.851 5.41 115. 0.845 5.50 114. 0.840
0.78 45. 0.721 1.30 44. 0.763 1.38 41. 0.812 1.46 36. 0.807 1.48 30. 0.754
1.29 54. 0.462 1.59 53. 0.830 1.66 49. 0.835 1.73 44. 0.814 1.75 38. 0.767
1.68 65. 0.665 1.97 64. 0.856 2.04 60. 0.849 2.10 55. 0.823 2.13 49. 0.781
1.88 71. 0.698 2.20 70. 0.864 2.28 67. 0.856 2.34 62. 0.829 2.36 56. 0.789
2.18 78. 0.761 2.46 77. 0.873 2.55 74. 0.863 2.62 69. 0.836 2.65 63. 0.798
2.14 86. 0.657 2.77 85. 0.881 2.88 81. 0.871 2.95 76. 0.845 2.98 71. 0.809
1.35 94. 0.218 3.10 93. 0.887 3.25 89. 0.879 3.35 85. 0.855 3.38 79. 0.820
2.78 102. 0.745 3.47 101. 0.891 3.67  98. 0.885 3.82  93. 0.864 3.86  88. 0.832
2.92 109. 0.723 3.88 108. 0.888 4.17 105. 0.884 4.36 101. 0.864 4.42  96. 0.834
3.62 115. 0.790 4.98 115. 0.856 5.26 115. 0.851 5.41 115. 0.845 5.50 114. 0.840
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2.78 102. 0.745 3.47 101. 0.891 3.67  98. 0.885 3.82  93. 0.864 3.86  88. 0.832
2.92 109. 0.723 3.88 108. 0.888 4.17 105. 0.884 4.36 101. 0.864 4.42  96. 0.834
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0.78 45. 0.721 1.30 44. 0.763 1.38 41. 0.812 1.46 36. 0.807 1.48 30. 0.754
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1.68 65. 0.665 1.97 64. 0.856 2.04 60. 0.849 2.10 55. 0.823 2.13 49. 0.781
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2.18 78. 0.761 2.46 77. 0.873 2.55 74. 0.863 2.62 69. 0.836 2.65 63. 0.798
2.14 86. 0.657 2.77 85. 0.881 2.88 81. 0.871 2.95 76. 0.845 2.98 71. 0.809
1.35 94. 0.218 3.10 93. 0.887 3.25 89. 0.879 3.35 85. 0.855 3.38 79. 0.820
2.78 102. 0.745 3.47 101. 0.891 3.67  98. 0.885 3.82  93. 0.864 3.86  88. 0.832
2.92 109. 0.723 3.88 108. 0.888 4.17 105. 0.884 4.36 101. 0.864 4.42  96. 0.834
3.62 115. 0.790 4.98 115. 0.856 5.26 115. 0.851 5.41 115. 0.845 5.50 114. 0.840
0.78 45. 0.721 1.30 44. 0.763 1.38 41. 0.812 1.46 36. 0.807 1.48 30. 0.754
1.29 54. 0.462 1.59 53. 0.830 1.66 49. 0.835 1.73 44. 0.814 1.75 38. 0.767
1.68 65. 0.665 1.97 64. 0.856 2.04 60. 0.849 2.10 55. 0.823 2.13 49. 0.781
1.88 71. 0.698 2.20 70. 0.864 2.28 67. 0.856 2.34 62. 0.829 2.36 56. 0.789
2.18 78. 0.761 2.46 77. 0.873 2.55 74. 0.863 2.62 69. 0.836 2.65 63. 0.798
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1.35 94. 0.218 3.10 93. 0.887 3.25 89. 0.879 3.35 85. 0.855 3.38 79. 0.820
2.78 102. 0.745 3.47 101. 0.891 3.67  98. 0.885 3.82  93. 0.864 3.86  88. 0.832
2.92 109. 0.723 3.88 108. 0.888 4.17 105. 0.884 4.36 101. 0.864 4.42  96. 0.834
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-5 0 5 10 20 30
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1.45 414. 0.880 1.60 407. 0.902 1.73 387. 0.884 1.82 358. 0.815 1.84 323. 0.709
1.49 432. 0.864 1.70 427. 0.897 1.84 410. 0.884 1.95 386. 0.826 1.97 357. 0.735
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1.04 299. 0.762 1.08 289. 0.766 1.18 262. 0.908 1.28 221. 0.841 1.32 172. 0.646
1.15 327. 0.841 1.22 318. 0.895 1.32 292. 0.908 1.42 253. 0.820 1.45 207. 0.659
1.21 343. 0.875 1.29 334. 0.908 1.40 309. 0.902 1.49 271. 0.813 1.52 226. 0.665
1.27 360. 0.882 1.37 352. 0.911 1.49 327. 0.895 1.58 290. 0.808 1.61 247. 0.671
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 1.19 327. 0.865 1.28 318. 0.906 1.42 293. 0.896 1.54 255. 0.786 1.58 210. 0.623
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 1.54 410. 0.833 1.83 406. 0.888 2.07 393. 0.884 2.24 374. 0.811 2.29 351. 0.721
 0.85 241. 0.750 0.89 230. 0.750 0.99 201. 0.870 1.13 157. 0.867 1.19 105. 0.546
 0.92 255. 0.845 0.96 245. 0.771 1.09 217. 0.864 1.23 176. 0.818 1.28 127. 0.552
 1.01 272. 0.722 1.07 263. 0.748 1.21 237. 0.906 1.35 198. 0.782 1.40 151. 0.561
 1.07 282. 0.717 1.13 273. 0.852 1.28 248. 0.901 1.42 209. 0.768 1.47 164. 0.565
 1.12 293. 0.809 1.19 284. 0.886 1.35 259. 0.892 1.50 222. 0.756 1.54 178. 0.568
 1.17 304. 0.846 1.26 296. 0.898 1.44 271. 0.882 1.58 235. 0.745 1.63 192. 0.570
 1.22 315. 0.858 1.33 307. 0.900 1.52 284. 0.873 1.67 248. 0.735 1.72 207. 0.570
 1.27 327. 0.860 1.41 319. 0.899 1.62 296. 0.866 1.78 262. 0.727 1.82 223. 0.571
 1.31 338. 0.852 1.49 331. 0.897 1.73 310. 0.861 1.89 278. 0.724 1.93 240. 0.576
 1.40 361. 0.810 1.66 355. 0.882 1.95 339. 0.866 2.13 314. 0.749 2.16 286. 0.623

-1
-1
INTAKE S1-2         D1-4          R100
COMPRE S2-3         D5-11         R101      V5   V6
PREMAS S3,17,4      D12-15                  V12
DUCTER S17,18       D16-19
NOZCON S18,19,1     D83           R103
ARITHY              D20-24        !(COPY)
COMPRE S4-5         D25-31        R102      V25
PREMAS S5,20,6      D32-35                  V32
DUCTER S20-21       D36-39
ARITHY              D40-46        !(ADD W1+W2)
COMPRE S6-7         D47-53        R104      V47  V48
PREMAS S7,22,8      D54-57
PREMAS S22,23,24    D58-61
BURNER S8-9         D62-64        R105      L1
MIXEES S9,23,10
TURBIN S10-11       D65-72,104,73              V66
MIXEES S11,24,12
TURBIN S12-13       D74-81,108,82              V75
MIXFUL S13,21,14    D86-88
DUCTER S14,15       D89-92       R106
NOZDIV S15,16,1     D93-94       R107
ARITHY              D120-126
ARITHY              D127-133
ARITHY              D134-140
ARITHY              D141-149
ARITHY              D150-158
ARITHY              D159-167
PLOTBD              D84,450,400,451,401,452,402,8
PERFOR S1,0,0       D95-98,107,100,105,103,0,106,0,0,0,0,0
CODEND

DATA ////
1 0.0                    !             ALTITUDE
2 0.0                     !INTAKE        I.S.A DEVIATION
3 0.0                     !              MACH NUMBER
4 -1.0                    !              PRESSURE RECOVERY

5 0.33                    !              Z=(R-Rc)/(Rs-Rc)
6 1                      !             PCN ROTATIONAL SPEED
7 1.7                    !COMPRE 1      DESIGN PRESSURE RATIO
8 0.89                     !              DESIGN ISENTROPIC EFFICIENCY
9 0.0                      !              ERROR SELECTION
10 1.0                     !               COMPRESSOR MAP NUMBER
11 0.0                     !               STATOR ANGLE

12 0.55                     !              BYPASS RATIO
13 0.0                     !PREMAS        LOSS IN W           ( BYPASS)
14 1.0                     !              PRESSURE RATIO
15 0.01                    !              LOSS IN PRESSURE

16 0.0                     !              SWITCH (NO REHEATING REQUIRED)
17 0.03                    !DUCTER        PRESSURE LOSS/INLET TOTAL PRESSURE
18 0.0                     !              COMBUSTION EFFICIENCY
19 1000000.0               !              LIMITING VALUE OF FUEL

20 5                       !              COPY         (SAME ROTATIONAL SPEED
21 -1                      !ARITHY        B.D.            FOR COMPRESSOR 1 ,2)
22 26                      !              B.D. ITEM NUMBER
23 -1                      !              B.D.
24 6.0                     !              B.D. ITEM NUMBER

25 0.87                     !              Z=(R-Rc)/(Rs-Rc)
26 1                       !              PCN ROTATIONAL SPEED
27 1.264                     !COMPRE 2      DESIGN PRESSURE RATIO
28 0.88                    !              DESIGN ISENTROPIC EFFICIENCY
29 1.0                     !              ERROR SELECTION
30 2.0                     !              COMPRESSOR MAP NUMBER
31 0.0                     !              STATOR ANGLE

32 0.02                  !             BYPASS RATIO
33 0.0                     !PREMAS        LOSS IN W           ( BYPASS)
34 1.0                     !              PRESSURE RATIO
```

```
35 0.01                  !          LOSS IN PRESSURE

36 0.0                   !          SWITCH (NO REHEATING REQUIRED)
37 0.03                  !DUCTER    PRESSURE LOSS/INLET TOTAL PRESSURE
38 0.0                   !          COMBUSTION EFFICIENCY
39 1000000.0             !          LIMITING VALUE OF FUEL

40 1                     !          ADDITION
41 -1                    !          BD.          {WC = WC1 + WC2 }
42 108                   !ARITHY    BD. ITEM NUMBER
43 -1                    !          BD.
44 101                   !          BD. ITEM NUMBER
45 -1                    !          BD.
46 102                   !          BD. ITEM NUMBER

47 0.8                   !          Z=(R-Rc)/(Rs-Rc)
48 1.0                   !          PCN ROTATIONAL SPEED
49 4                    !COMPRE 3 DESIGN PRESSURE RATIO
50 0.88                  !          DESIGN ISENTROPIC EFFICIENCY
51 1.0                   !          ERROR SELECTION
52 3.0                   !          COMPRESSOR MAP NUMBER
53 0.0                   !          STATOR ANGLE

54 0.15                  !          BYPASS RATIO
55 0.0                   !PREMAS    LOSS IN W          ( COOLING )
56 1.0                   !          PRESSURE RATIO
57 0.0                   !          LOSS IN PRESSURE

58 0.6                   !          BYPASS RATIO
59 0.0                   !PREMAS    LOSS IN W          ( COOLING )
60 1.0                   !          PRESSURE RATIO
61 0.0                   !          LOSS IN PRESSURE

62 0.05                  !          TOTAL PRESSURE LOSS
63 0.99                   !BURNER    COMBUSTION
64 -1.0                  !          FUEL FLOW

65 100000.0              !          AUXILIARY WORK
66 -1.0                  !      RELATIVE NO-DIMENSIONAL INLET MASS FLOW
67 -1.0                  !TURBIN 1  RELATIVE NON-DIMENSIONAL SPEED CN
68 0.9                   !          DESIGN ISENTROPIC EFFICIENCY
69 -1.0                  !          RELATIVE ROTATIONAL SPEED
70 3.0                   !          COMPRESSOR NUMBER
71 4.0                   !          TURBINE MAP NUMBER
72 -1.0                  !          POWER LAW INDEX
73 0.0                   !          NGV ANGLE

74 0.0                   !          AUXILIARY WORK
75 -1.0                  !      RELATIVE NO-DIMENSIONAL INLET MASS FLOW
76 0.6                    !          RELATIVE NON-DIMENSIONAL SPEED CN
77 0.9                   !TURBIN 2  DESIGN ISENTROPIC EFFICIENCY
78 -1.0                  !          RELATIVE ROTATIONAL SPEED
79 1.0                   !          COMPRESSOR NUMBER
80 4.0                   !          TURBINE MAP NUMBER
81 -1.0                  !          POWER LAW INDEX
82 0.0                   !          NGV  ANGLE

83 -1.0                   !        NUMBER OF COMPRESSOR PROVIDING STREAM 2

84 1.0                   !MIXFUL    MACH NUMBER IN BD(3)
85 0.42                  !          MACH NUMBER

86 2.0                   !      NUMBER OF COMPRESSOR PROVIDING STREAM 2
87 1.0                   !MIXFUL    MACH NUMBER IN BD(3)
88 0.42                  !          MACH NUMBER

89 1.0                   !          SWITCH (NO REHEATING REQUIRED)
90 0.03                  !DUCTER    PRESSURE LOSS/INLET TOTAL PRESSURE
91 0.91                  !          COMBUSTION EFFICIENCY
92 1000000.0             !          LIMITING VALUE OF FUEL

93 -1.0                  !NOZDIV    FOR AREAS
94 -1.0                  !          THROAT AREA

95 -1.0                  !          POWER FOR POWER TURBINE
96 -1.0                  !PERFOR    PROPELLER EFFICIENCY
97 0.0                   !          SCALING INDEX
98 181500.0              !          REQUIRED DESIGN-POINT NET THRUST

120 4
121 -1
122 400                  !NDMF LPC
123 3
124 4
125 2
126 4

127 4
128 -1
129 401
130 5
131 4                   !NDMF IPC
132 4
```

```
133 4

134 4
135 -1
136 402
137 7
138 4
139 6
140 4                    !NDMF HPC

141 29
142 -1
143 450
144 3
145 2
146 2
147 6
148 2
149 4

150 29
151 -1
152 451
153 4
154 2
155 4
156 6
157 4
158 4

159 29
160 -1
161 452
162 6
163 2
164 6
165 6
166 6
167 4

-1
1 2 445
9 6 1240
-1
93 -2
1 460
3 0.5
84 2
-1
9 6 1480
-1
84 3
1 2290
3 0.6
-1
9 6 1390
-1
1 9150
3 0.95
84 4
-1
9 6 1150
-1
89 2
94 0.93
84 5
-1
9 6 1395
13 8 0.96
19 8 0.4
21 8 0.15
15 6 1525
-1
94 1.1
11 5
31 0
1 9750
3 1.05
84 6
-1
9 6 1390
19 8 0.3
21 8 0.2
15 6 1500
-1
84 7
94 1.07
11 10
-1
9 6 1365
-1
11 15
84 8
```

```
94 1.02
-1
-1
11 20
84 9
94 0.97
-1
9 6 1375
15 6 1700
-1
11 25
94 0.95
31 -5
84 10
1 12250
3 1.6
-1
9 6 1500
19 8 0.15
21 8 0.25
15 6 1650
-1
94 1.03
1 18750
11 30
3 2.7
31 -10
93 -1
84 11
-1
15 6 1280
9 6 1800
19 8 0.015
21 8 0.164
16 8 3
-1
84 12
94 1.05
89 1
-1
19 8 0.007
21 8 0.175
-1
-3
```

TURBOMATCH OUTPUT FILE

```
Double  Bypass  Engine


  TURBOMATCH SCHEME-VAX VERSION (11/11/94)

 LIMITS:100 Codewords, 800 Brick Data Items, 50 Station Vector
 15 BD Items printable by any call of:-
 OUTPUT, OUTPBD, OUTPSV, PLOTIT, PLOTBD or PLOTSV


      ***** DESIGN POINT ENGINE CALCULATIONS *****


 ***** AMBIENT AND INLET PARAMETERS *****
 Alt. =       0.0      I.S.A. Dev. =     0.000       Mach No. =  0.00
 Etar = 1.0000         Momentum Drag =       0.00

 ***** COMPRESSOR  1 PARAMETERS *****
 PRSF =  0.99263E+00     ETASF =  0.10190E+01     WASF =  0.99278E+00
 Z = 0.33000             PR =    1.700            ETA = 0.89000
 PCN =   1.0000          CN = 1.00000             COMWK =  0.23694E+08
 STATOR ANGLE =    0.00

 ***** CONVERGENT NOZZLE  1 PARAMETERS *****
 Area =  0.6732          Exit Velocity =  303.96  Gross Thrust =  72860.59
 Nozzle Coeff. =  0.97939E+00

 ***** COMPRESSOR  2 PARAMETERS *****
 PRSF =  0.10185E+01     ETASF =  0.99624E+00     WASF =  0.10175E+01
 Z = 0.87000             PR =    1.264            ETA = 0.88000
 PCN =   1.0000          CN = 1.00000             COMWK =  0.53980E+07
```

STATOR ANGLE = 0.00

***** COMPRESSOR 3 PARAMETERS *****
PRSF = 0.10240E+01 ETASF = 0.99421E+00 WASF = 0.10051E+01
Z = 0.80000 PR = 4.000 ETA = 0.88000
PCN = 1.0000 CN = 1.00000 COMWK = 0.39951E+08
STATOR ANGLE = 0.00

***** COMBUSTION CHAMBER PARAMETERS *****
ETASF = 0.99000E+00
ETA = 0.99000 DLP = 0.4298 WFB = 3.2116

***** TURBINE 1 PARAMETERS *****
CNSF = 0.71084E+02 ETASF = 0.10567E+01 TFSF = 0.52242E+00
DHSF = 0.10480E+05
TF = 414.346 ETA = 0.90000 CN = 2.060
AUXWK = 0.10000E+06 NGV ANGLE = 0.00

***** TURBINE 2 PARAMETERS *****
CNSF = 0.64728E+02 ETASF = 0.10567E+01 TFSF = 0.25276E+00
DHSF = 0.86384E+04
TF = 414.346 ETA = 0.90000 CN = 2.060
AUXWK = 0.00000E+00 NGV ANGLE = 0.00

***** MIXING MACH NUMBERS *****
Station 13, M= 0.421 Station 21,M = 0.439 Station 14,M = 0.422

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA = 0.9100 DLP = 0.0619 WFB = 0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
Throat Area = 0.7286 Throat Velocity = 530.66 Throat Mach No. = 1.0000
Exit Area = 0.7308 Exit Velocity = 557.59 Exit Mach No. = 1.0599
Nozzle Coeff. = 0.9808 Gross Thrust = 111273.23

Nozzle exit and throat areas are fixed.
AREA RATIO (EXIT/THROAT): 1.00

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	445.000	1.00000	1.00000	288.15	288.15	0.0	******
2	0.00000	445.000	******	1.00000	******	288.15	******	******
3	0.00000	445.000	******	1.70000	******	341.14	******	******
4	0.00000	200.250	******	1.70000	******	341.14	******	******
5	0.00000	200.250	******	2.14880	******	367.88	******	******
6	0.00000	196.245	******	2.14880	******	367.88	******	******
7	0.00000	196.245	******	8.59520	******	566.66	******	******
8	0.00000	166.808	******	8.59520	******	566.66	******	******
9	0.01925	170.020	******	8.16544	******	1250.00	******	******
10	0.01741	187.682	******	8.16544	******	1190.72	******	******
11	0.01741	187.682	******	3.82317	******	1011.87	******	******
12	0.01637	199.457	******	3.82317	******	987.30	******	******
13	0.01637	199.457	1.83601	2.06338	836.08	861.06	238.5	1.0789
14	0.01604	203.462	1.83426	2.06234	827.18	852.06	238.0	1.0921
15	0.01604	203.462	******	2.00047	******	852.06	******	******
16	0.01604	203.462	1.00000	2.00047	713.35	852.06	557.6	0.7308
17	0.00000	244.750	******	1.71000	******	341.14	******	******
18	0.00000	244.750	******	1.65870	******	341.14	******	******
19	0.00000	244.750	1.00000	1.65870	295.17	341.14	304.0	0.6732
20	0.00000	4.005	******	2.15880	******	367.88	******	******
21	0.00000	4.005	1.83601	2.09404	354.30	367.88	165.5	0.0132
22	0.00000	29.437	******	8.59520	******	566.66	******	******
23	0.00000	17.662	******	8.59520	******	566.66	******	******
24	0.00000	11.775	******	8.59520	******	566.66	******	******

Gross Thrust = 184133.83
Momentum Drag = 0.00
Net Thrust = 184133.83
Fuel Flow = 3.2116
s.f.c. = 17.44169
Sp. Thrust = 413.784
Time Now 11:38:41

****
1 9150
3 0.95
-1
9 6 1150
-1
Time Now 11:38:42

****

***** OFF DESIGN ENGINE CALCULATIONS. Converged after 8 Loops *****

***** AMBIENT AND INLET PARAMETERS *****

```
Alt. =   9150.0        I.S.A. Dev. =     0.000       Mach No. =  0.95
Etar = 1.0000          Momentum Drag =  69600.40

***** COMPRESSOR  1 PARAMETERS *****
PRSF =  0.99263E+00       ETASF =  0.10190E+01      WASF =  0.99278E+00
Z = 0.31403               PR =   1.671              ETA = 0.89173
PCN =   0.9572            CN = 0.98864              COMWK =  0.11621E+08
STATOR ANGLE =    0.00

***** CONVERGENT NOZZLE  1 PARAMETERS *****
Area =  0.6732            Exit Velocity =  326.40  Gross Thrust =  53903.41
Nozzle Coeff. =  0.97487E+00

***** COMPRESSOR  2 PARAMETERS *****
PRSF =  0.10185E+01       ETASF =  0.99624E+00      WASF =  0.10175E+01
Z = 0.86925               PR =   1.257              ETA = 0.88097
PCN =   0.9572            CN = 0.99134              COMWK =  0.26281E+07
STATOR ANGLE =    0.00

***** COMPRESSOR  3 PARAMETERS *****
PRSF =  0.10240E+01       ETASF =  0.99421E+00      WASF =  0.10051E+01
Z = 0.80002               PR =   3.955              ETA = 0.88010
PCN =   0.9602            CN = 0.99533              COMWK =  0.19681E+08
STATOR ANGLE =    0.00

***** COMBUSTION CHAMBER PARAMETERS *****
ETA = 0.99000             DLP = 0.2223              WFB =   1.5329

***** TURBINE  1 PARAMETERS *****
CNSF =  0.71084E+02       ETASF =  0.10567E+01      TFSF =  0.52242E+00
DHSF =  0.10480E+05
TF =  414.615              ETA = 0.90001             CN =  2.062
AUXWK =  0.10000E+06      NGV ANGLE = 0.00

***** TURBINE  2 PARAMETERS *****
CNSF =  0.64728E+02       ETASF =  0.10567E+01      TFSF =  0.25276E+00
DHSF =  0.86384E+04
TF =  414.743              ETA = 0.89980             CN =  2.060
AUXWK =  0.00000E+00      NGV ANGLE = 0.00

***** MIXING MACH NUMBERS *****
Station 13, M= 0.421     Station 21,M = 0.478    Station 14,M = 0.422

***** DUCT/AFTER BURNING 1 PARAMETERS *****
ETA =  0.9100             DLP = 0.0315              WFB =   0.0000

***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
Throat Area =  0.7286    Throat Velocity =  507.72    Throat Mach No. =  1.0000
Exit Area =  0.8397      Exit Velocity =  685.86      Exit Mach No. =  1.4589
Nozzle Coeff. =  0.9782  Gross Thrust =  72915.92

Nozzle exit area floating/throat area fixed.
AREA RATIO (EXIT/THROAT): 1.15


Scale Factor on above Mass Flows, Areas, Thrusts & Powers =     1.0000
```

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	241.568	0.29663	0.53043	228.68	270.10	288.1	******
2	0.00000	241.568	******	0.53043	******	270.10	******	******
3	0.00000	241.568	******	0.88659	******	318.03	******	******
4	0.00000	107.207	******	0.88659	******	318.03	******	******
5	0.00000	107.207	******	1.11470	******	342.40	******	******
6	0.00000	104.839	******	1.11470	******	342.40	******	******
7	0.00000	104.839	******	4.40809	******	526.66	******	******
8	0.00000	89.113	******	4.40809	******	526.66	******	******
9	0.01720	90.646	******	4.18581	******	1150.00	******	******
10	0.01555	100.081	******	4.18581	******	1095.61	******	******
11	0.01555	100.081	******	1.94656	******	927.02	******	******
12	0.01462	106.372	******	1.94656	******	904.78	******	******
13	0.01462	106.372	0.93243	1.04788	763.41	786.65	228.8	1.0789
14	0.01431	108.677	0.93209	1.04889	754.54	777.84	228.2	1.0921
15	0.01431	108.677	******	1.01743	******	777.84	******	******
16	0.01431	108.677	0.29663	1.01743	561.84	777.84	685.9	0.8397
17	0.00000	134.517	******	0.89659	******	318.03	******	******
18	0.00000	134.517	******	0.86970	******	318.03	******	******
19	0.00000	134.517	0.45936	0.86970	264.92	318.03	326.4	0.6732
20	0.00000	2.305	******	1.12470	******	342.40	******	******
21	0.00000	2.305	0.93320	1.09096	327.48	342.40	173.2	0.0132
22	0.00000	15.726	******	4.40809	******	526.66	******	******
23	0.00000	9.435	******	4.40809	******	526.66	******	******
24	0.00000	6.290	******	4.40809	******	526.66	******	******

```
  Gross Thrust = 126819.33
Momentum Drag =  69600.40
   Net Thrust =  57218.93
    Fuel Flow =    1.5329
       s.f.c. =  26.78959
   Sp. Thrust =   236.864
```

```
************************************************
11 30
 31 -10
 93 -1
 94 1.01
 89 1
 -1
 16 8 2.9
 19 8 0.01
 21 8 0.17
 -1
                                             Time Now 11:38:44

 ***********************************************
      ***** OFF DESIGN ENGINE CALCULATIONS. Converged after  11 Loops *****


 ***** AMBIENT AND INLET PARAMETERS *****
 Alt. =  18750.0       I.S.A. Dev. =    0.000       Mach No. =  2.70
 Etar = 0.8465         Momentum Drag = 186870.45

 ***** VARIABLE COMPRESSOR  1 PARAMETERS *****
 PRSF =  0.99263E+00      ETASF =  0.10190E+01     WASF =  0.99278E+00
 Z = 0.95248              PR =   1.778             ETA = 0.68316
 PCN =   1.3497           CN = 0.99493             COMWK =  0.32580E+08
 STATOR ANGLE =  30.00

 ***** CONVERGENT NOZZLE  1 PARAMETERS *****
 Area =  0.0100         Exit Velocity =  470.11  Gross Thrust =   2508.50
 Nozzle Coeff. =  0.80176E+00

 ***** VARIABLE COMPRESSOR  2 PARAMETERS *****
 PRSF =  0.10185E+01      ETASF =  0.99624E+00     WASF =  0.10175E+01
 Z = 0.22654              PR =   1.172             ETA = 0.83923
 PCN =   1.3497           CN = 0.96842             COMWK =  0.84649E+07
 STATOR ANGLE = -10.00

 ***** COMPRESSOR  3 PARAMETERS *****
 PRSF =  0.10240E+01      ETASF =  0.99421E+00     WASF =  0.10051E+01
 Z = 0.87412              PR =   3.180             ETA = 0.88050
 PCN =   1.2668           CN = 0.92040             COMWK =  0.49703E+08
 STATOR ANGLE =   0.00

 ***** COMBUSTION CHAMBER PARAMETERS *****
 ETA = 0.99000            DLP = 0.5011             WFB =   3.5918

 ***** TURBINE  1 PARAMETERS *****
 CNSF =  0.71084E+02      ETASF =  0.10567E+01     TFSF =  0.52242E+00
 DHSF =  0.10480E+05
 TF =  416.352             ETA = 0.90560            CN =  2.166
 AUXWK =  0.10000E+06      NGV ANGLE = 0.00

 ***** TURBINE  2 PARAMETERS *****
 CNSF =  0.64728E+02      ETASF =  0.10567E+01     TFSF =  0.25276E+00
 DHSF =  0.86384E+04
 TF =  425.991             ETA = 0.91331            CN =  2.294
 AUXWK =  0.00000E+00      NGV ANGLE = 0.00

 ***** MIXING MACH NUMBERS *****
 Station 13, M= 0.587    Station 21,M = 0.953    Station 14,M = 0.645

 ***** DUCT/AFTER BURNING 1 PARAMETERS *****
 ETA =  0.9100            DLP = 0.0587             WFB =   0.0000

 ARATIO =  2.871     Out of Range: use Data for   2.150
 ***** CONVERGENT/DIVERGENT NOZZLE 1 PARAMETERS *****
 Throat Area =  1.0100    Throat Velocity =  600.04    Throat Mach No. =  1.0000
 Exit Area =  2.9000      Exit Velocity = 1148.84      Exit Mach No. =  2.5618
 Nozzle Coeff. =  0.9546  Gross Thrust = 266201.56

 Nozzle exit and throat areas are fixed.
 AREA RATIO (EXIT/THROAT): 2.87
 Nozzle is Underexpanded.
```

Scale Factor on above Mass Flows, Areas, Thrusts & Powers = 1.0000

Station	F.A.R.	Mass Flow	Pstatic	Ptotal	Tstatic	Ttotal	Vel	Area
1	0.00000	234.456	0.06577	1.53692	216.65	530.28	797.0	******
2	0.00000	234.456	******	1.30097	******	530.28	******	******
3	0.00000	234.456	******	2.31337	******	662.64	******	******
4	0.00000	230.867	******	2.31337	******	662.64	******	******
5	0.00000	230.867	******	2.71036	******	696.91	******	******
6	0.00000	161.821	******	2.71036	******	696.91	******	******
7	0.00000	161.821	******	8.61999	******	974.54	******	******
8	0.00000	137.548	******	8.61999	******	974.54	******	******
9	0.02611	141.140	******	8.11890	******	1800.00	******	******
10	0.02361	155.704	******	8.11890	******	1728.60	******	******
11	0.02361	155.704	******	3.73692	******	1477.98	******	******
12	0.02220	165.413	******	3.73692	******	1450.34	******	******
13	0.02220	165.413	1.48083	1.84209	1188.24	1250.68	392.0	0.9600
14	0.01556	234.479	1.50104	1.95626	1028.63	1096.93	402.8	1.1300
15	0.01556	234.479	******	1.89757	******	1096.93	******	******
16	0.01556	234.479	0.09660	1.89757	509.37	1096.93	1148.8	2.9000
17	0.00000	3.590	******	2.32337	******	662.64	******	******
18	0.00000	3.590	******	2.25367	******	662.64	******	******
19	0.00000	3.590	1.20621	2.25367	557.74	662.64	470.1	0.0100
20	0.00000	69.067	******	2.72036	******	696.91	******	******
21	0.00000	69.067	1.48061	2.63875	596.09	696.91	462.3	0.1700
22	0.00000	24.273	******	8.61999	******	974.54	******	******
23	0.00000	14.564	******	8.61999	******	974.54	******	******
24	0.00000	9.709	******	8.61999	******	974.54	******	******

Gross Thrust = 268710.06
Momentum Drag = 186870.45
Net Thrust = 81839.62
Fuel Flow = 3.5918
s.f.c. = 43.88809
Sp. Thrust = 349.061
Time Now 11:38:44

****
-3

INPUT FILE FOR THE L.P. COMPRESSOR

For Compdes Program

Double Bypass Engine.
Design point is The Take-Off Point.

```
LP COMPRESSOR DESIGN        !TITLE
2                                    !NOMBER OF STAGE
3                                    !ANNULUS GEOMETRY SELECTOR
1                                    !IGV's SELECTION
0.35                                 !HUB/TIP RATIO
88                                   !ISENTROPIC EFFICIENCY
200 200 200 200        !INLET AXIAL VELOCITY
26.49 28.48                 !TEMP. RISE PER STAGE
288.15                               !INLET TOTAL TEMPERATURE
1                                    !INLET TOTAL PRESSURE
1.72                                 !DESIGN PRESSURE RATIO
445                                  !INLET MASS FLOW
1                                    !VORTEX DESIGN SELECTOR
0 0 0 0                     !STAGE INLET SWIRL ANGLE
4500                                 !ROTATIONAL SPEED
0.98 0.93                   !STAGE WORK DONE FACTORS
0.8 0.7 0.8 0.7             !S/C RATIO
3 2.75                      !ROTOR H/C RATIO
0.8 0.7                     !BMH REACTION
0.01 0.01                   !ROTOR TIP CLEARANCE
0.00001                     !EQUIVALENT SAND ROUGHNESS
```

INPUT FILE FOR THE LP. COMPRESSOR

For Compdes Program

Double Bypass Engine.
Design point is The Take-Off Point.

```
LP COMPRESSOR DESIGN        !TITLE
1                                    !NOMBER OF STAGE
3                                    !ANNULUS GEOMETRY SELECTOR
1                                    !IGV's SELECTION
0.7                                  !HUB/TIP RATIO
88                                   !ISENTROPIC EFFICIENCY
170 170 170                 !INLET AXIAL VELOCITY
25.57                                !TEMP. RISE PER STAGE
335.21                               !INLET TOTAL TEMPERATURE
1.6                                  !INLET TOTAL PRESSURE
1.25                                 !DESIGN PRESSURE RATIO
200.25                      !INLET MASS FLOW
1                                    !VORTEX DESIGN SELECTOR
0 5 0                                !STAGE INLET SWIRL ANGLE
4500                                 !ROTATIONAL SPEED
0.98                                 !STAGE WORK DONE FACTORS
0.8 0.7                     !S/C RATIO
3.5                                  !ROTOR H/C RATIO
0.8                                  !BMH REACTION
0.01                                 !ROTOR TIP CLEARANCE
0.00001                     !EQUIVALENT SAND ROUGHNESS
```

INPUT FILE FOR THE H.P. COMPRESSOR

For Compdes Program

Double Bypass Engine.
Design point is The Take-Off Point.

```
H.P COMPRESSOR DESIGN       !TITLE
5                                    !NOMBER OF STAGE
3                                    !ANNULUS GEOMETRY SELECTOR
0                                    !IGV's SELECTION
0.55                                 !HUB/TIP RATIO
88                                   !ISENTROPIC EFFICIENCY
200 200 200 200 200 200 200!INLET AXIAL VELOCITY
35.8 37.8 43.8 41.8 39.8   !TEMP. RISE PER STAGE
368.58                               !INLET TOTAL TEMPERATURE
2.15                                 !INLET TOTAL PRESSURE
```

```
4                                                  !DESIGN PRESSURE RATIO
196.245                                  !INLET MASS FLOW
1                                                  !VORTEX DESIGN SELECTOR
25 25 25 25 25 25 25      !STAGE INLET SWIRL ANGLE
7000                                               !ROTATIONAL SPEED
0.99 0.95 0.92 0.92 0.9       !STAGE WORK DONE FACTORS
0.8 0.75 0.8 0.75 0.8 0.75 0.8 0.75 0.8 0.75  !S/C RATIO
2 2 2 2        2                                   !ROTOR H/C RATIO
0.82 0.71 0.69 0.69 0.69      !BMH REACTION
0.01 0.01 0.01 0.01 0.01      !ROTOR TIP CLEARANCE
0.00001                                  !EQUIVALENT SAND ROUGHNESS
```

OUTPUT FILE FOR THE L.P. COMPRESSOR

From Compdes Program

Double-Bypass Engine

```
PROPERTY DISTRIBUTION THROUGH A 2 STAGE AXIAL COMPRESSOR WITH IGVs
--------------------------------------------------------
            CONSTANT OUTER RADIUS TYPE
            --------------------------

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)-------------445.00
OVERALL PRESSURE RATIO--------- 1.72
INLET AXIAL VELOCITY (m/s)-----200.00
ISENTROPIC EFFICIENCY (%)------88.00
OVERALL TEMPERATURE RISE (K)---- 54.97
INLET TOTAL TEPERATURE  (K)---288.15
INLET TOTAL PRESSURE  (atm)---- 1.00
ROTATIONAL SPEED (rpm)------ 4500.00
INLET IGV'S STA 1 STA 2 STA 3 STA 4 STA 5 STA 6 STA 7 STA 8 STA 9
TOTAL TEMPERATURE RISE (K)
        26.49 28.48 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL TEMPERATURE (K)
288.15 288.15 314.64 343.12 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AVERAGE TOTAL TEMPERATURE (K)
        301.39 328.88 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET STATIC TEMPERATURE (K)
268.21 268.21 294.73 323.25 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL PRESSURE (atm)
1.000 1.000 1.316 1.726 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET STATIC PRESSURE (atm)
0.778 0.778 1.047 1.400 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET DENSITY (kg/cubic m)
1.0245 1.0245 1.2541 1.5292 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET SPECIFIC HEAT (kJ/kg K)
1.0030 1.0030 1.0045 1.0067 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

AVERAGE SPECIFIC HEAT (kJ/kg K)
        1.0037 1.0055 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET ANNULUS AREA (square m)
2.1718 2.1718 1.7742 1.4550 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET INNER RADIUS (cm)
31.07 31.07 47.23 56.98 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET OUTER RADIUS (cm)
88.76 88.76 88.76 88.76 0.00 0.00 0.00 0.00 0.00 0.00 0.00

BLADE HEIGHT (cm)
57.69 57.69 41.53 31.78 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET HUB/TIP RATIO
0.350 0.350 0.532 0.642 0.000 0.000 0.000 0.000 0.000 0.000 0.000

ABSOLUTE OUTLET VELOCITY (m/s)
200.00 200.00 200.00 200.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET SWIRL ANGLE (DEGREES)
0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AXIAL VELOCITY (m/s)
200.00 200.00 200.00 200.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

PRESSURE RATIO
        1.3161 1.3114 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

ENTHALPY RISE (kJ/kg)
        26.588 28.638 0.000 0.000 0.000 0.000 0.000 0.000 0.000
```

COMPRESSOR POWER (kW)--------- 24575.461

OVERALL PRESSURE RATIO-------- 1.7260

L.P COMPRESSOR DESIGN ITITLE

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT OUTER RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9	
OUTSIDE RADIUS	88.76	88.76	88.76	0.00	0.00	0.00	0.00	0.00	0.00	0.00
HUB RADIUS	31.07	47.23	56.98	0.00	0.00	0.00	0.00	0.00	0.00	0.00
INLET HEIGHT	57.693	41.528	31.780	0.000	0.000	0.000	0.000	0.000	0.000	0.000
INLET HUB/TIP RATIO	0.3500	0.5321	0.6420	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	
ASPECT RATIO	3.0000	2.7500	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	
MID CHORD HEIGHT-ROTOR	55.002	40.031	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
MID CHORD HEIGHT-STATOR	46.746	35.069	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-ROTOR	18.334	14.557	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-STATOR	15.582	12.752	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL CHORD	12.826	9.265	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL SPACE	5.500	4.367	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL CHORD,	15.519	12.701	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL SPACE	4.675	3.826	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TOTAL STAGE LENGTH	38.520	30.158	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
NUMBER OF ROTOR BLADES	26.2	37.1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	
NUMBER OF STATOR BLADES	37.7	50.1	0.0	0.0	0.0	0.0	0.0	0.0	0.0	

TOTAL COMPRESSOR LENGTH 68.678

STAGE 1

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	152.7	217.5	282.3	347.1	411.9	M/S
ABS. STATOR OUTLET VELOCITY (V0)	200.0	200.0	200.0	200.0	200.0	M/S
REL. ROTOR INLET VELOCITY (V1)	251.6	295.5	346.0	400.6	457.9	M/S
REL. ROTOR OUTLET VELOCITY (V2)	201.5	220.5	273.3	335.2	399.7	M/S
ABS. STATOR INLET VELOCITY (V3)	267.5	235.7	221.9	214.7	210.6	M/S
ABS. STATOR OUTLET VELOCITY (V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	0.678	0.591	0.496	0.402	0.317	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	0.849	0.937	0.973	0.992	1.002	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.77	0.90	1.05	1.22	1.39	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.80	0.69	0.65	0.63	0.61	
ROTOR DECELERATION (V2/V1)	0.80	0.75	0.79	0.84	0.87	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.36	0.44	0.38	0.30	0.24	
STATOR DECELERATION (V0/V3)	0.75	0.85	0.90	0.93	0.95	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	1.16	0.57	0.34	0.23	0.16	
FLOW COEFFICENT (VA/U)	1.31	0.92	0.71	0.58	0.49	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	0.0	0.0	0.0	0.0	0.0	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	37.4	47.4	54.7	60.1	64.1	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	-7.1	24.9	43.0	53.4	60.0	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	41.6	31.9	25.7	21.3	18.2	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION	44.5	22.5	11.7	6.7	4.1	DEGREES
STATOR DEFLECTION	41.6	31.9	25.7	21.3	18.2	DEGREES
REACTION	41.8	71.3	83.0	88.7	92.0	PERCENT
SPACE CHORD RATIO (ROTOR)	0.610	0.747	0.800	0.821	0.830	
SPACE CHORD RATIO (STATOR)	0.411	0.558	0.700	0.839	0.977	
DIFFUSION FACTOR (ROTOR)	0.414	0.411	0.321	0.243	0.187	
DIFFUSION FACTOR (STATOR)	0.389	0.299	0.250	0.221	0.203	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.659	1.675	1.517	1.397	1.320	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.665	1.500	1.428	1.389	1.366	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00335	0.02973	0.05545	0.10851	
SHOCK LOSS COEFFICIENT (STATOR)	0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.23E+06	.24E+06	.26E+06	.28E+06	.29E+06	
REYNLDS NUMBER (STATOR)	.30E+06	.32E+06	.33E+06	.34E+06	.34E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.02649	0.01944	0.02170	0.02537	0.02949
PROFILE LOSS COEFFICIENT (STATOR)	0.03494	0.02873	0.02399	0.02033	0.01769
SECONDARY LOSS COEFFICIENT (ROTOR)	0.00684				
SECONDARY LOSS COEFFICIENT (STATOR)	0.00972				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00814				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00589				
TOTAL LOSS COEFFICIENT (ROTOR)	0.07889				
TOTAL LOSS COEFFICIENT (STATOR)	0.04075				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.012

ROTOR ISENTROPIC EFFICIENCY 86.29 PERCENT

STAGE ISENTROPIC EFFICIENCY 82.88 PERCENT

STAGE PRESSURE RATIO 1.29

STAGE 2

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	226.1	273.3	320.4	367.6	414.7	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	200.0	200.0	200.0	200.0	200.0	M/S
REL. ROTOR INLET VELOCITY	(V1)	301.9	338.6	377.7	418.5	460.4	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	219.3	256.5	300.5	347.2	394.9	M/S
ABS. STATOR INLET VELOCITY	(V3)	242.0	229.6	221.9	216.8	213.3	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	0.783	0.696	0.607	0.522	0.443	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	1.220	1.260	1.285	1.300	1.311	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.88	0.98	1.10	1.22	1.34	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.68	0.64	0.62	0.61	0.59	
ROTOR DECELERATION	(V2/V1)	0.73	0.76	0.80	0.83	0.86	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	0.47	0.43	0.37	0.31	0.26	
STATOR DECELERATION	(V0/V3)	0.83	0.87	0.90	0.92	0.94	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	0.60	0.41	0.30	0.23	0.18	
FLOW COEFFICENT	(VA/U)	0.88	0.73	0.62	0.54	0.48	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	0.0	0.0	0.0	0.0	0.0	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	48.5	53.8	58.0	61.4	64.3	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	24.2	38.8	48.3	54.8	59.6	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	34.3	29.4	25.7	22.7	20.4	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION		24.3	15.0	9.7	6.6	4.7	DEGREES
STATOR DEFLECTION		34.3	29.4	25.7	22.7	20.4	DEGREES
REACTION		69.9	79.4	85.0	88.6	91.0	PERCENT
SPACE CHORD RATIO (ROTOR)		0.747	0.783	0.800	0.808	0.811	
SPACE CHORD RATIO (STATOR)		0.512	0.607	0.700	0.792	0.884	
DIFFUSION FACTOR (ROTOR)		0.442	0.373	0.306	0.251	0.208	
DIFFUSION FACTOR (STATOR)		0.318	0.278	0.250	0.231	0.216	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.729	1.603	1.490	1.407	1.346	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.531	1.467	1.428	1.401	1.382	
SHOCK LOSS COEFFICIENT (ROTOR)		0.00259	0.00616	0.03325	0.05303	0.08613	
SHOCK LOSS COEFFICIENT (STATOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)		.23E+06	.24E+06	.26E+06	.27E+06	.28E+06	
REYNLDS NUMBER (STATOR)		.36E+06	.37E+06	.38E+06	.39E+06	.39E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.01962	0.02066	0.02348	0.02671	0.03004
PROFILE LOSS COEFFICIENT (STATOR)	0.03053	0.02659	0.02356	0.02114	0.01917
SECONDARY LOSS COEFFICIENT (ROTOR)	0.00630				
SECONDARY LOSS COEFFICIENT (STATOR)	0.00972				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00983				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00643				
TOTAL LOSS COEFFICIENT (ROTOR)	0.07647				
TOTAL LOSS COEFFICIENT (STATOR)	0.04035				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.008

ROTOR ISENTROPIC EFFICIENCY 85.85 PERCENT

STAGE ISENTROPIC EFFICIENCY 82.70 PERCENT

STAGE PRESSURE RATIO 1.29

****

OVERALL PRESSURE RATIO 1.66

COMPRESSOR ISENTROPIC EFFICIENCY 82.09 PERCENT

COMPRESSOR POLYTROPIC EFFICIENCY 83.33 PERCENT

OUTPUT FILE FOR THE I.P. COMPRESSOR

From Compdes Program
Double-Bypass Engine

PROPERTY DISTRIBUTION THROUGH A 1 STAGE AXIAL COMPRESSOR WITH IGVs

CONSTANT OUTER RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)--------------200.25
OVERALL PRESSURE RATIO--------- 1.25
INLET AXIAL VELOCITY (m/s)-----170.00
ISENTROPIC EFFICIENCY (%)-------88.00
OVERALL TEMPERATURE RISE (K)---- 25.57
INLET TOTAL TEPERATURE (K)----335.21
INLET TOTAL PRESSURE (atm)---- 1.60
ROTATIONAL SPEED (rpm)------- 4500.00
INLET IGV'S STA 1 STA 2 STA 3 STA 4 STA 5 STA 6 STA 7 STA 8 STA 9
TOTAL TEMPERATURE RISE (K)
25.57 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL TEMPERATURE (K)
335.21 335.21 360.78 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AVERAGE TOTAL TEMPERATURE (K)
347.99 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET STATIC TEMPERATURE (K)
320.85 320.74 346.45 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET TOTAL PRESSURE (atm)
1.600 1.600 2.012 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET STATIC PRESSURE (atm)
1.372 1.371 1.745 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000

OUTLET DENSITY (kg/cubic m)
1.5100 1.5087 1.7778 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET SPECIFIC HEAT (kJ/kg K)
1.0060 1.0060 1.0084 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

AVERAGE SPECIFIC HEAT (kJ/kg K)
1.0071 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET ANNULUS AREA (square m)
0.7801 0.7807 0.6626 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

OUTLET INNER RADIUS (cm)
48.84 48.82 52.53 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET OUTER RADIUS (cm)
69.78 69.78 69.78 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

BLADE HEIGHT (cm)
20.93 20.95 17.24 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET HUB/TIP RATIO
0.700 0.700 0.753 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000

ABSOLUTE OUTLET VELOCITY (m/s)
170.00 170.65 170.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

OUTLET SWIRL ANGLE (DEGREES)
0.00 5.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

AXIAL VELOCITY (m/s)
170.00 170.00 170.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00 0.00

PRESSURE RATIO
1.2573 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000 0.0000

ENTHALPY RISE (kJ/kg)
25.752 0.000 0.000 0.000 0.000 0.000 0.000 0.000 0.000

COMPRESSOR POWER (kW)---------- 5156.922

OVERALL PRESSURE RATIO--------- 1.2573

I.P COMPRESSOR DESIGN ITITLE

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT OUTER RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

STAGE 1 STAGE 2 STAGE 3 STAGE 4 STAGE 5 STAGE 6 STAGE 7 STAGE 8 STAGE 9

OUTSIDE RADIUS	69.78	69.78	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
HUB RADIUS	48.82	52.53	0.00	0.00	0.00	0.00	0.00	0.00	0.00	0.00
INLET HEIGHT	20.955	17.243	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000
INLET HUB/TIP RATIO	0.6997	0.7529	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	
ASPECT RATIO	3.5000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	0.0000	
MID CHORD HEIGHT-ROTOR	20.377	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
MID CHORD HEIGHT-STATOR	18.497	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-ROTOR	5.822	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TRUE CHORD-STATOR	5.285	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL CHORD	3.859	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
ROTOR AXIAL SPACE	1.747	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL CHORD,	5.210	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
STATOR AXIAL SPACE	1.586	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
TOTAL STAGE LENGTH	12.401	0.000	0.000	0.000	0.000	0.000	0.000	0.000	0.000	
NUMBER OF ROTOR BLADES	80.4	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	
NUMBER OF STATOR BLADES	102.8	0.0	0.0	0.0	0.0	0.0	0.0	0.0	0.0	

TOTAL COMPRESSOR LENGTH 12.401

STAGE 1

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	231.4	255.4	279.4	303.4	327.5	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	170.9	170.8	170.6	170.6	170.5	M/S
REL. ROTOR INLET VELOCITY	(V1)	272.9	293.4	314.5	335.9	357.7	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	197.2	217.9	240.8	264.9	289.6	M/S
ABS. STATOR INLET VELOCITY	(V3)	214.9	207.6	201.9	197.4	193.7	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	170.0	170.0	170.0	170.0	170.0	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	170.0	170.0	170.0	170.0	170.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	170.0	170.0	170.0	170.0	170.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	1.091	1.031	0.971	0.909	0.849	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	1.601	1.627	1.646	1.661	1.674	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.76	0.82	0.88	0.94	1.00	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.58	0.56	0.55	0.53	0.52	
ROTOR DECELERATION	(V2/V1)	0.72	0.74	0.77	0.79	0.81	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	0.48	0.45	0.41	0.38	0.34	
STATOR DECELERATION	(V0/V3)	0.79	0.82	0.84	0.86	0.88	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	0.49	0.40	0.34	0.29	0.25	
FLOW COEFFICENT	(VA/U)	0.73	0.67	0.61.	0.56	0.52	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	6.0	5.5	5.0	4.6	4.3	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	51.5	54.6	57.3	59.6	61.6	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	30.4	38.7	45.1	50.1	54.1	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	37.7	35.0	32.6	30.5	28.7	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	0.0	0.0	0.0	0.0	0.0	DEGREES
ROTOR DEFLECTION		21.0	15.9	12.2	9.5	7.6	DEGREES
STATOR DEFLECTION		37.7	35.0	32.6	30.5	28.7	DEGREES
REACTION		67.7	73.5	77.9	81.2	83.9	PERCENT
SPACE CHORD RATIO (ROTOR)		0.791	0.798	0.800	0.799	0.797	
SPACE CHORD RATIO (STATOR)		0.583	0.642	0.700	0.758	0.815	
DIFFUSION FACTOR (ROTOR)		0.442	0.397	0.354	0.314	0.280	
DIFFUSION FACTOR (STATOR)		0.363	0.340	0.321	0.305	0.291	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.723	1.641	1.566	1.501	1.447	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.634	1.592	1.560	1.535	1.515	
SHOCK LOSS COEFFICIENT (ROTOR)		0.00000	0.00000	0.00256	0.00456	0.00659	
SHOCK LOSS COEFFICIENT (STATOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)		.14E+06	.15E+06	.15E+06	.16E+06	.16E+06	
REYNLDS NUMBER (STATOR)		.21E+06	.21E+06	.22E+06	.22E+06	.22E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.02463	0.02579	0.02817	0.03037	0.03298
PROFILE LOSS COEFFICIENT (STATOR)	0.02930	0.02702	0.02555	0.02360	0.02223
SECONDARY LOSS COEFFICIENT (ROTOR)	0.00836				
SECONDARY LOSS COEFFICIENT (STATOR)	0.01540				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00715				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00469				
TOTAL LOSS COEFFICIENT (ROTOR)	0.04665				
TOTAL LOSS COEFFICIENT (STATOR)	0.04563				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.004

ROTOR ISENTROPIC EFFICIENCY 92.58 PERCENT

STAGE ISENTROPIC EFFICIENCY 89.21 PERCENT

STAGE PRESSURE RATIO 1.26

OVERALL PRESSURE RATIO 1.26

COMPRESSOR ISENTROPIC EFFICIENCY 89.21 PERCENT

COMPRESSOR POLYTROPIC EFFICIENCY 89.56 PERCENT

OUTPUT FILE FOR THE H.P. COMPRESSOR

From Compdes Program

Double-Bypass Engine

PROPERTY DISTRIBUTION THROUGH A 5 STAGE AXIAL COMPRESSOR WITHOUT IGVs

CONSTANT OUTER RADIUS TYPE

THE FOLLOWING CONSTANT DATA HAS BEEN USED FOR THIS RUN:

MASS FLOW (kg/s)--------------196.24
OVERALL PRESSURE RATIO---------- 4.00
INLET AXIAL VELOCITY (m/s)-----200.00
ISENTROPIC EFFICIENCY (%)-------88.00
OVERALL TEMPERATURE RISE (K)---199.00
INLET TOTAL TEPERATURE (K)---368.58
INLET TOTAL PRESSURE (atm)--- 2.15
ROTATIONAL SPEED (rpm)------ 7000.00
INLET IGV'S STA 1 STA 2 STA 3 STA 4 STA 5 STA 6 STA 7 STA 8 STA 9
TOTAL TEMPERATURE RISE (K)
35.80 37.80 43.80 41.80 39.80 0.00 0.00 0.00 0.00

OUTLET TOTAL TEMPERATURE (K)
368.58 368.58 404.38 442.18 485.98 527.78 567.58 0.00 0.00 0.00 0.00

AVERAGE TOTAL TEMPERATURE (K)
386.48 423.28 464.08 506.88 547.68 0.00 0.00 0.00 0.00

OUTLET STATIC TEMPERATURE (K)
344.45 344.45 380.35 418.28 462.25 504.24 544.23 0.00 0.00 0.00 0.00

OUTLET TOTAL PRESSURE (atm)
2.150 2.150 2.889 3.846 5.214 6.815 8.644 0.000 0.000 0.000 0.000

OUTLET STATIC PRESSURE (atm)
1.695 1.695 2.327 3.158 4.360 5.782 7.420 0.000 0.000 0.000 0.000

OUTLET DENSITY (kg/cubic m)
1.7368 1.7368 2.1601 2.6652 3.3299 4.0482 4.8135 0.0000 0.0000 0.0000 0.0000

OUTLET SPECIFIC HEAT (kJ/kg K)
1.0092 1.0092 1.0134 1.0188 1.0263 1.0344 1.0429 0.0000 0.0000 0.0000 0.0000

AVERAGE SPECIFIC HEAT (kJ/kg K)
1.0112 1.0160 1.0224 1.0302 1.0385 0.0000 0.0000 0.0000 0.0000

OUTLET ANNULUS AREA (square m)
0.5650 0.5650 0.4543 0.3682 0.2947 0.2424 0.2038 0.0000 0.0000 0.0000 0.0000

OUTLET INNER RADIUS (cm)
27.93 27.93 33.65 37.50 40.50 42.50 43.92 0.00 0.00 0.00 0.00

OUTLET OUTER RADIUS (cm)
50.78 50.78 50.78 50.78 50.78 50.78 50.78 0.00 0.00 0.00 0.00

BLADE HEIGHT (cm)
22.85 22.85 17.13 13.28 10.28 8.27 6.85 0.00 0.00 0.00 0.00

OUTLET HUB/TIP RATIO
0.550 0.550 0.663 0.739 0.798 0.837 0.865 0.000 0.000 0.000 0.000

ABSOLUTE OUTLET VELOCITY (m/s)
220.68 220.68 220.68 220.68 220.68 220.68 220.68 0.00 0.00 0.00 0.00

OUTLET SWIRL ANGLE (DEGREES)
25.00 25.00 25.00 25.00 25.00 25.00 25.00 0.00 0.00 0.00 0.00

AXIAL VELOCITY (m/s)
200.00 200.00 200.00 200.00 200.00 200.00 200.00 0.00 0.00 0.00 0.00

PRESSURE RATIO
1.3438 1.3313 1.3558 1.3070 1.2683 0.0000 0.0000 0.0000 0.0000

ENTHALPY RISE (kJ/kg)
36.200 38.405 44.782 43.063 41.334 0.000 0.000 0.000 0.000

COMPRESSOR POWER (kW)---------- 39991.367

OVERALL PRESSURE RATIO-------- 4.0203

H.P COMPRESSOR DESIGN ITITLE

AXIAL COMPRESSOR/FAN "FREE VORTEX" DESIGN

CONSTANT OUTER RADIUS TYPE

ALL ANNULUS DIMENSIONS IN CENTIMETRES

	STAGE 1	STAGE 2	STAGE 3	STAGE 4	STAGE 5	STAGE 6	STAGE 7	STAGE 8	STAGE 9	
OUTSIDE RADIUS	50.78	50.78	50.78	50.78	50.78	50.78	0.00	0.00	0.00	0.00
HUB RADIUS	27.93	33.65	37.50	40.50	42.50	43.92	0.00	0.00	0.00	0.00
INLET HEIGHT	22.849	17.127	13.275	10.276	8.271	6.852	0.000	0.000	0.000	0.000
INLET HUB/TIP RATIO	0.5500	0.6627	0.7386	0.7976	0.8371	0.8651	0.0000	0.0000	0.0000	
ASPECT RATIO	2.0000	2.0000	2.0000	2.0000	2.0000	0.0000	0.0000	0.0000	0.0000	
MID CHORD HEIGHT-ROTOR	21.681	16.364	12.680	9.892	8.006	0.000	0.000	0.000	0.000	
MID CHORD HEIGHT-STATOR	18.767	14.406	11.155	8.875	7.287	0.000	0.000	0.000	0.000	
TRUE CHORD-ROTOR	10.840	8.182	6.340	4.946	4.003	0.000	0.000	0.000	0.000	
TRUE CHORD-STATOR	9.383	7.203	5.577	4.438	3.644	0.000	0.000	0.000	0.000	
ROTOR AXIAL CHORD	9.661	7.003	5.375	4.039	3.191	0.000	0.000	0.000	0.000	
ROTOR AXIAL SPACE	3.252	2.455	1.902	1.484	1.201	0.000	0.000	0.000	0.000	
STATOR AXIAL CHORD,	7.928	6.059	4.586	3.693	3.052	0.000	0.000	0.000	0.000	
STATOR AXIAL SPACE	2.815	2.161	1.673	1.331	1.093	0.000	0.000	0.000	0.000	
TOTAL STAGE LENGTH	23.655	17.677	13.536	10.548	8.537	0.000	0.000	0.000	0.000	
NUMBER OF ROTOR BLADES	28.9	40.9	55.0	72.8	91.8	0.0	0.0	0.0	0.0	
NUMBER OF STATOR BLADES	37.0	50.7	67.9	87.5	108.4	0.0	0.0	0.0	0.0	

TOTAL COMPRESSOR LENGTH 73.953

STAGE 1

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	209.0	248.7	288.5	328.2	367.9	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	237.8	227.4	220.7	216.1	212.9	M/S
REL. ROTOR INLET VELOCITY	(V1)	215.5	244.5	279.5	317.2	356.2	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	221.3	200.1	211.4	241.2	279.6	M/S
ABS. STATOR INLET VELOCITY	(V3)	363.6	324.2	297.3	278.2	264.1	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	220.7	220.7	220.7	220.7	220.7	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	1.708	1.607	1.479	1.338	1.195	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	1.556	1.783	1.935	2.040	2.115	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.58	0.66	0.75	0.85	0.96	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.99	0.86	0.78	0.73	0.69	
ROTOR DECELERATION	(V2/V1)	1.03	0.82	0.76	0.76	0.78	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	-.05	0.33	0.43	0.42	0.38	
STATOR DECELERATION	(V0/V3)	0.61	0.68	0.74	0.79	0.84	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	0.84	0.59	0.44	0.34	0.27	
FLOW COEFFICENT	(VA/U)	0.96	0.80	0.69	0.61	0.54	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	32.8	28.4	25.0	22.3	20.1	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	21.9	35.1	44.3	50.9	55.8	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	-25.3	-1.8	18.9	34.0	44.3	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	56.6	51.9	47.7	44.0	40.8	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	30.0	27.3	25.0	23.0	21.3	DEGREES
ROTOR DEFLECTION		47.2	36.9	25.4	16.9	11.5	DEGREES
STATOR DEFLECTION		26.6	24.6	22.7	21.0	19.4	DEGREES
REACTION		-3.4	27.0	45.7	58.0	66.6	PERCENT
SPACE CHORD RATIO (ROTOR)		0.639	0.761	0.800	0.797	0.780	
SPACE CHORD RATIO (STATOR)		0.503	0.627	0.750	0.872	0.992	
DIFFUSION FACTOR (ROTOR)		0.233	0.410	0.425	0.380	0.324	
DIFFUSION FACTOR (STATOR)		0.514	0.461	0.418	0.381	0.351	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.377	1.647	1.690	1.615	1.522	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.900	1.772	1.686	1.624	1.579	
SHOCK LOSS COEFFICIENT (ROTOR)		0.00000	0.00000	0.00000	0.00175	0.00523	
SHOCK LOSS COEFFICIENT (STATOR)		0.00626	0.00211	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)		.43E+06	.39E+06	.39E+06	.41E+06	.43E+06	
REYNLDS NUMBER (STATOR)		.36E+06	.36E+06	.37E+06	.37E+06	.38E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		0.955	0.980	1.000	1.015	1.028	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.03498	0.01996	0.01769	0.01837	0.02125
PROFILE LOSS COEFFICIENT (STATOR)	0.02465	0.02135	0.01901	0.01656	0.01502
SECONDARY LOSS COEFFICIENT (ROTOR)			0.01419		
SECONDARY LOSS COEFFICIENT (STATOR)			0.01247		
ANNULUS LOSS COEFFICIENT (ROTOR)			0.00880		
ANNULUS LOSS COEFFICIENT (STATOR)			0.00927		
TOTAL LOSS COEFFICIENT (ROTOR)			0.04683		
TOTAL LOSS COEFFICIENT (STATOR)			0.04273		

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.003

ROTOR ISENTROPIC EFFICIENCY 95.29 PERCENT

STAGE ISENTROPIC EFFICIENCY 90.74 PERCENT

STAGE PRESSURE RATIO 1.35

STAGE 2

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	249.5	279.4	309.4	339.4	369.4	M/S
ABS. STATOR OUTLET VELOCITY (V0)	231.0	225.1	220.7	217.3	214.7	M/S
REL. ROTOR INLET VELOCITY (V1)	240.6	266.5	294.5	323.6	353.3	M/S
REL. ROTOR OUTLET VELOCITY (V2)	202.0	202.5	217.5	241.5	270.3	M/S
ABS. STATOR INLET VELOCITY (V3)	342.3	318.5	300.2	285.8	274.2	M/S
ABS. STATOR OUTLET VELOCITY (V4)	220.7	220.7	220.7	220.7	220.7	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	2.239	2.119	1.987	1.848	1.707	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	2.359	2.531	2.660	2.760	2.839	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.62	0.68	0.75	0.83	0.90	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.87	0.80	0.75	0.71	0.68	
ROTOR DECELERATION (V2/V1)	0.84	0.76	0.74	0.75	0.77	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.30	0.42	0.45	0.44	0.41	
STATOR DECELERATION (V0/V3)	0.64	0.69	0.74	0.77	0.80	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	0.65	0.52	0.42	0.35	0.30	
FLOW COEFFICENT (VA/U)	0.80	0.72	0.65	0.59	0.54	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	30.0	27.3	25.0	23.0	21.3	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	33.8	41.4	47.2	51.8	55.5	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	-8.0	9.0	23.2	34.1	42.3	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	54.2	51.1	48.2	45.6	43.2	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	28.6	26.7	25.0	23.5	22.2	DEGREES
ROTOR DEFLECTION	41.8	32.4	24.1	17.7	13.2	DEGREES
STATOR DEFLECTION	25.7	24.4	23.2	22.1	21.0	DEGREES
REACTION	21.1	37.2	48.8	57.4	64.0	PERCENT
SPACE CHORD RATIO (ROTOR)	0.749	0.792	0.800	0.791	0.776	
SPACE CHORD RATIO (STATOR)	0.573	0.662	0.750	0.838	0.925	
DIFFUSION FACTOR (ROTOR)	0.413	0.455	0.439	0.399	0.355	
DIFFUSION FACTOR (STATOR)	0.491	0.457	0.428	0.402	0.380	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.639	1.733	1.715	1.648	1.572	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.835	1.761	1.704	1.660	1.624	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00000	0.00000	0.00095	0.00348	
SHOCK LOSS COEFFICIENT (STATOR)	0.00245	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.39E+06	.38E+06	.39E+06	.40E+06	.42E+06	
REYNLDS NUMBER (STATOR)	.38E+06	.39E+06	.39E+06	.39E+06	.40E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	0.969	0.986	1.000	1.012	1.022	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.02167	0.01790	0.01739	0.01842	0.02060
PROFILE LOSS COEFFICIENT (STATOR)	0.02292	0.02058	0.01859	0.01690	0.01567
SECONDARY LOSS COEFFICIENT (ROTOR)			0.01420		
SECONDARY LOSS COEFFICIENT (STATOR)			0.01305		
ANNULUS LOSS COEFFICIENT (ROTOR)			0.00906		
ANNULUS LOSS COEFFICIENT (STATOR)			0.00922		
TOTAL LOSS COEFFICIENT (ROTOR)			0.04334		
TOTAL LOSS COEFFICIENT (STATOR)			0.04169		

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.002

ROTOR ISENTROPIC EFFICIENCY 95.49 PERCENT

STAGE ISENTROPIC EFFICIENCY 91.18 PERCENT

STAGE PRESSURE RATIO 1.34

STAGE 3

	BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED (U)	277.1	300.3	323.6	346.8	370.0	M/S
ABS. STATOR OUTLET VELOCITY (V0)	227.7	223.8	220.7	218.1	216.0	M/S
REL. ROTOR INLET VELOCITY (V1)	261.3	282.7	305.0	327.8	351.0	M/S
REL. ROTOR OUTLET VELOCITY (V2)	200.1	203.5	215.4	232.9	254.2	M/S
ABS. STATOR INLET VELOCITY (V3)	347.8	330.1	315.3	302.8	292.2	M/S
ABS. STATOR OUTLET VELOCITY (V4)	220.7	220.7	220.7	220.7	220.7	M/S
AXIAL VELOCITY AT ROTOR INLET (VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET (VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET (PS1)	2.939	2.812	2.677	2.538	2.397	ATM
STATIC PRESSURE AT ROTOR OUTLET (PS2)	3.317	3.483	3.618	3.730	3.824	ATM
REL. ROTOR INLET MACH NUMBER (M1REL)	0.64	0.69	0.75	0.80	0.86	
ABS. STATOR INLET MACH NUMBER (M3ABS)	0.84	0.79	0.76	0.72	0.69	
ROTOR DECELERATION (V2/V1)	0.77	0.72	0.71	0.71	0.72	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT (DP/D)	0.41	0.48	0.50	0.50	0.48	
STATOR DECELERATION (V0/V3)	0.63	0.67	0.70	0.73	0.76	STATOR DE HALLER NUMBER
STAGE LOADING (DELTA H/U SQUARED)	0.63	0.54	0.46	0.40	0.36	
FLOW COEFFICENT (VA/U)	0.72	0.67	0.62	0.58	0.54	
ABS. STATOR AIR OUTLET ANGLE (ALPHA0)	28.6	26.7	25.0	23.5	22.2	DEGREES
REL. ROTOR AIR INLET ANGLE (ALPHA1)	40.1	45.0	49.0	52.4	55.3	DEGREES
REL. ROTOR AIR OUTLET ANGLE (ALPHA2)	-2.1	10.7	21.8	30.8	38.1	DEGREES
ABS. STATOR AIR INLET ANGLE (ALPHA3)	54.9	52.7	50.6	48.7	46.8	DEGREES
ABS. STATOR AIR OUTLET ANGLE (ALPHA4)	27.6	26.2	25.0	23.9	22.8	DEGREES

ROTOR DEFLECTION	42.2	34.3	27.3	21.6	17.1	DEGREES
STATOR DEFLECTION	27.3	26.5	25.6	24.8	24.0	DEGREES
REACTION	29.0	39.6	47.9	54.7	60.2	PERCENT
SPACE CHORD RATIO (ROTOR)	0.784	0.800	0.800	0.790	0.777	
SPACE CHORD RATIO (STATOR)	0.615	0.683	0.750	0.817	0.884	
DIFFUSION FACTOR (ROTOR)	0.497	0.510	0.491	0.459	0.421	
DIFFUSION FACTOR (STATOR)	0.521	0.499	0.479	0.461	0.444	
EQUIVALENT DIFFUSION FACTOR (ROTOR)	1.783	1.831	1.810	1.754	1.686	
EQUIVALENT DIFFUSION FACTOR (STATOR)	1.898	1.843	1.798	1.760	1.728	
SHOCK LOSS COEFFICIENT (ROTOR)	0.00000	0.00000	0.00000	0.00000	0.00193	
SHOCK LOSS COEFFICIENT (STATOR)	0.00145	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)	.39E+06	.39E+06	.40E+06	.41E+06	.42E+06	
REYNLDS NUMBER (STATOR)	.40E+06	.40E+06	.40E+06	.40E+06	.40E+06	
AXIAL VELOCITY RATIO (ROTOR)	1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)	0.978	0.990	1.000	1.009	1.017	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.01915	0.01733	0.01708	0.01776	0.01888
PROFILE LOSS COEFFICIENT (STATOR)	0.02144	0.01975	0.01824	0.01696	0.01590
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01773				
SECONDARY LOSS COEFFICIENT (STATOR)	0.01608				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00871				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00900				
TOTAL LOSS COEFFICIENT (ROTOR)	0.04487				
TOTAL LOSS COEFFICIENT (STATOR)	0.04382				

EFFICIENCY LOSS DUE TO TIP CLEARANCE 0.002

ROTOR ISENTROPIC EFFICIENCY 95.68 PERCENT

STAGE ISENTROPIC EFFICIENCY 91.40 PERCENT

STAGE PRESSURE RATIO 1.36

STAGE 4

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	298.3	316.4	334.5	352.7	370.8	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	225.7	223.0	220.7	218.7	217.0	M/S
REL. ROTOR INLET VELOCITY	(V1)	278.4	295.7	313.4	331.4	349.5	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	203.4	211.9	224.2	239.3	256.4	M/S
ABS. STATOR INLET VELOCITY	(V3)	329.2	317.5	307.2	298.2	290.3	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	220.7	220.7	220.7	220.7	220.7	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	3.974	3.841	3.703	3.561	3.418	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	4.741	4.875	4.990	5.088	5.174	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.65	0.69	0.73	0.77	0.81	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.76	0.73	0.70	0.68	0.66	
ROTOR DECELERATION	(V2/V1)	0.73	0.72	0.72	0.72	0.73	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	0.47	0.49	0.49	0.48	0.46	
STATOR DECELERATION	(V0/V3)	0.67	0.70	0.72	0.74	0.76	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	0.53	0.47	0.42	0.38	0.34	
FLOW COEFFICENT	(VA/U)	0.67	0.63	0.60	0.57	0.54	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	27.6	26.2	25.0	23.9	22.8	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	44.1	47.4	50.3	52.9	55.1	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	10.4	19.3	26.9	33.3	38.7	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	52.6	50.9	49.4	47.9	46.4	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	27.0	26.0	25.0	24.1	23.2	DEGREES
ROTOR DEFLECTION		33.7	28.2	23.5	19.6	16.4	DEGREES
STATOR DEFLECTION		25.6	25.0	24.4	23.8	23.2	DEGREES
REACTION		38.6	45.5	51.2	56.1	60.3	PERCENT
SPACE CHORD RATIO (ROTOR)		0.803	0.805	0.800	0.791	0.780	
SPACE CHORD RATIO (STATOR)		0.648	0.699	0.750	0.801	0.851	
DIFFUSION FACTOR (ROTOR)		0.496	0.485	0.463	0.436	0.407	
DIFFUSION FACTOR (STATOR)		0.484	0.468	0.452	0.438	0.425	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.805	1.795	1.760	1.712	1.661	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.813	1.778	1.748	1.722	1.698	
SHOCK LOSS COEFFICIENT (ROTOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
SHOCK LOSS COEFFICIENT (STATOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)		.41E+06	.41E+06	.42E+06	.43E+06	.44E+06	
REYNLDS NUMBER (STATOR)		.44E+06	.44E+06	.44E+06	.45E+06	.45E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		0.983	0.992	1.000	1.007	1.014	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.01739	0.01708	0.01742	0.01821	0.01926
PROFILE LOSS COEFFICIENT (STATOR)	0.02081	0.01948	0.01831	0.01730	0.01629
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01512				
SECONDARY LOSS COEFFICIENT (STATOR)	0.01446				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00930				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00911				

TOTAL LOSS COEFFICIENT (ROTOR)	0.04229
TOTAL LOSS COEFFICIENT (STATOR)	0.04201

EFFICIENCY LOSS DUE TO TIP CLEARANCE	0.001
ROTOR ISENTROPIC EFFICIENCY	95.57 PERCENT
STAGE ISENTROPIC EFFICIENCY	91.45 PERCENT
STAGE PRESSURE RATIO	1.31

STAGE 5

		BLADE ROOT	25%	BLADE MID HEIGHT	75%	BLADE TIP	
BLADE SPEED	(U)	312.5	327.2	341.9	356.6	371.2	M/S
ABS. STATOR OUTLET VELOCITY	(V0)	224.5	222.5	220.7	219.1	217.7	M/S
REL. ROTOR INLET VELOCITY	(V1)	290.4	304.6	319.1	333.7	348.5	M/S
REL. ROTOR OUTLET VELOCITY	(V2)	209.9	219.1	230.4	243.2	257.1	M/S
ABS. STATOR INLET VELOCITY	(V3)	319.3	310.7	303.0	296.0	289.7	M/S
ABS. STATOR OUTLET VELOCITY	(V4)	220.7	220.7	220.7	220.7	220.7	M/S
AXIAL VELOCITY AT ROTOR INLET	(VA1)	200.0	200.0	200.0	200.0	200.0	M/S
AXIAL VELOCITY AT ROTOR OUTLET	(VA2)	200.0	200.0	200.0	200.0	200.0	M/S
STATIC PRESSURE AT ROTOR INLET	(PS1)	5.220	5.082	4.941	4.796	4.650	ATM
STATIC PRESSURE AT ROTOR OUTLET	(PS2)	6.338	6.454	6.557	6.649	6.730	ATM
REL. ROTOR INLET MACH NUMBER	(M1REL)	0.65	0.68	0.71	0.75	0.78	
ABS. STATOR INLET MACH NUMBER	(M3ABS)	0.70	0.68	0.67	0.65	0.63	
ROTOR DECELERATION	(V2/V1)	0.72	0.72	0.72	0.73	0.74	ROTOR DE HALLER NUMBER
PRESSURE RISE COEFFICENT	(DP/D)	0.48	0.48	0.48	0.47	0.46	
STATOR DECELERATION	(V0/V3)	0.69	0.71	0.73	0.75	0.76	STATOR DE HALLER NUMBER
STAGE LOADING	(DELTA H/U SQUARED)	0.47	0.43	0.39	0.36	0.33	
FLOW COEFFICENT	(VA/U)	0.64	0.61	0.58	0.56	0.54	
ABS. STATOR AIR OUTLET ANGLE	(ALPHA0)	27.0	26.0	25.0	24.1	23.2	DEGREES
REL. ROTOR AIR INLET ANGLE	(ALPHA1)	46.5	49.0	51.2	53.2	55.0	DEGREES
REL. ROTOR AIR OUTLET ANGLE	(ALPHA2)	17.6	24.1	29.7	34.7	38.9	DEGREES
ABS. STATOR AIR INLET ANGLE	(ALPHA3)	51.2	49.9	48.7	47.5	46.3	DEGREES
ABS. STATOR AIR OUTLET ANGLE	(ALPHA4)	25.0	25.0	25.0	25.0	25.0	DEGREES
ROTOR DEFLECTION		28.8	24.9	21.4	18.5	16.0	DEGREES
STATOR DEFLECTION		26.2	24.9	23.7	22.5	21.3	DEGREES
REACTION		43.9	48.8	53.1	56.9	60.2	PERCENT
SPACE CHORD RATIO (ROTOR)		0.809	0.806	0.800	0.792	0.783	
SPACE CHORD RATIO (STATOR)		0.674	0.712	0.750	0.788	0.825	
DIFFUSION FACTOR (ROTOR)		0.482	0.467	0.446	0.424	0.401	
DIFFUSION FACTOR (STATOR)		0.464	0.451	0.438	0.426	0.414	
EQUIVALENT DIFFUSION FACTOR (ROTOR)		1.788	1.764	1.730	1.690	1.650	
EQUIVALENT DIFFUSION FACTOR (STATOR)		1.803	1.760	1.722	1.686	1.654	
SHOCK LOSS COEFFICIENT (ROTOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
SHOCK LOSS COEFFICIENT (STATOR)		0.00000	0.00000	0.00000	0.00000	0.00000	
REYNLDS NUMBER (ROTOR)		.43E+06	.43E+06	.44E+06	.45E+06	.46E+06	
REYNLDS NUMBER (STATOR)		.48E+06	.48E+06	.49E+06	.49E+06	.49E+06	
AXIAL VELOCITY RATIO (ROTOR)		1.000	1.000	1.000	1.000	1.000	
AXIAL VELOCITY RATIO (STATOR)		1.000	1.000	1.000	1.000	1.000	

CASCADE PREDICTION METHOD FOR ESTIMATING EFFICIENCY

PROFILE LOSS COEFFICIENT (ROTOR)	0.01704	0.01719	0.01769	0.01830	0.01936
PROFILE LOSS COEFFICIENT (STATOR)	0.01982	0.01902	0.01829	0.01753	0.01700
SECONDARY LOSS COEFFICIENT (ROTOR)	0.01378				
SECONDARY LOSS COEFFICIENT (STATOR)	0.01361				
ANNULUS LOSS COEFFICIENT (ROTOR)	0.00967				
ANNULUS LOSS COEFFICIENT (STATOR)	0.00918				
TOTAL LOSS COEFFICIENT (ROTOR)	0.04137				
TOTAL LOSS COEFFICIENT (STATOR)	0.04112				

EFFICIENCY LOSS DUE TO TIP CLEARANCE	0.001
ROTOR ISENTROPIC EFFICIENCY	95.34 PERCENT
STAGE ISENTROPIC EFFICIENCY	91.21 PERCENT
STAGE PRESSURE RATIO	1.27

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OVERALL PRESSURE RATIO	4.09
COMPRESSOR ISENTROPIC EFFICIENCY	88.13 PERCENT
COMPRESSOR POLYTROPIC EFFICIENCY	90.15 PERCENT

APPENDIX E

COMPRESSOR & TURBINE DESIGN PARAMETERS

Compressor Aerodynamic And Mechanical Parameters

A- Inlet Axial velocity

The inlet axial velocity in a multistage axial-flow compressor depends on the application. For instance, the inlet axial velocity for an industrial gas turbine is of the order of 150 m/s, whereas for advanced aero engines it can be in excess of 200 m/s. The latter higher values of axial velocity are used to provide the high flow rate per unit frontal area required for turbojet and turbofan engines. On the other hand, exit axial velocity is usually fixed by the diffuser and combuster requirements.

B- Inlet Mach Number

For a subsonic compressor cascade at zero incidence, the inlet 'critical' and 'maximum' Mach numbers are in the range of 0.7 and 0.85. High Mach number narrows the effective operating incidence range and can lead to poor off-design performance. However, with the development of high speed profiles (thin blade sections of quite different shape) such as double circular arc and controlled diffusion airfoils, a stage can be designed to operate efficiently with a relative Mach number up to 1.2 over part of the blade height. This is called a transonic stage. However with fans of large bypass ratio, the inlet relative Mach number at the rotor tip may be of the order of 1.5. The use of IGVs will slightly reduce the inlet relative Mach number.

C- Hup-Tip Ratio

The inlet hub-tip ratio normally varies between 0.4 and 0.6 while for the outlet no more than 0.92 is used. Using a very low hub-tip ratio increases the mismatch between compressor and turbine diameters. In addition, both mechanical and aerodynamic design of the first stage is more complex. For a fixed limited blade speed, the rotational speed is a function of the hub-tip ratio. There is therefore a wide range of solutions, and the final choice will depend on the experience of the particular designer.

D- Tip Speed

The centrifugal stress in the rotor blade is determined by the tip speed, blade material, and hub-tip ratio. Once blade material and hub-tip ratio are selected the centrifugal stress is only a function of tip speed. In fact, the first stage is the longest and most highly stressed. The blades are subject to fluctuating gas bending stresses which add to the centrifugal stress and may cause fatigue failure. Normally, in compressor design the tip speed is of the order of 350 m/s. Up to this value, stress problems are not usually critical in the sizing of the annulus. However, tip speeds around 450 m/s are commonly used in fans of high bypass ratio turbofans with low hub-tip ratio.

E- Stage Loading and Flow Coefficient

The choice of a realistic stage loading is the single most important decision in the design process. The wrong choice may lead to untold problems later with little possibility of reaching the combination of efficiency, pressure ratio, mass flow, and operating range originally intended. However. fundamental to all of the aerodynamics design are the basic decisions of an aerodynamic nature. At blade mid-height a choice must be made for the local flow coefficient, $\phi = V_a / U$ and the stage loading coefficient $\Psi = \Delta H/U^2$, where V_a is the axial velocity, U is the blade speed, and ΔH is the stagnation enthalpy rise.

The flow coefficient effectively determines the performance of the stage. The axial velocity used may be the local value or the mean value across the annulus. As ϕ is reduced at the off-design conditions, the incidence rises bringing changes in the blade operation. Most axial-flow compressors are designed with ϕ between 0.3 and 0.6. and the lower end of this range is more common nowadays. Figure 4.1 shows the relationship between these coefficients.

F- Degree Of Reaction

The degree of reaction is a useful concept in compressor design. It relates to the ratio of static pressure rise in the rotor to that in the entire stage. It can also be defined in terms of enthalpy rise. The degree of reaction can range from 0% to 100%. The limiting values are of particular interest. For zero reaction, the rotor blades are of impulse type and all the static pressure rise occurs in the stator. For 100% reaction, the stators are of impulse type. Therefore, the 50% reaction is attractive since both components of the stage share the diffusion equally.

G- Stage Temperature Rise

In different compressor designs, the stage temperature rise can vary widely, depending on the application. For instance, it may vary from 10K to 30 K for subsonic stages and may be 45K or even higher in high-performance transonic stages. The temperature rise is not normally considered constant for each stage through the compressor. It is normal to design for lower temperature rises in the first and last stages. The first stage, for example, has a high Mach number at the tip of the rotor and the inlet flow may become distorted with significant variations in axial velocity across the compressor annulus. By reducing the temperature rise, these problems can be partially alleviated. For the last stage, the outlet flow goes into the diffuser at entry of the combustion chamber. It would be desirable to have a purely axial velocity at exit. Therefore, the swirl should be kept as low as possible implying lower stage loading. The choice of the number of stages is directly related to the temperature rise chosen for each stage and individual blade loading. As tip speed increases, the temperature rise is higher for the same stage loading so that the number of stages is reduced.

H- Diffusion Factor

The blade loading limits are usually assessed by the level of diffusion facto (or equivalent diffusion ratio). Essentially, the diffusion factor defines the ratio of the peak velocity on the suction surface of the blade to that at the trailing edge. Additionally, the flow turning is accounted for. Another method used for assessing the allowable diffusion is the de Haller number. This defines the ratio of the blade inlet to outlet velocity, V_2/V_1. Values of the de Haller number are usually > o.72. This is equivalent to a static pressure ratio rise (ΔP) not exceeding about 0.5 times the dynamic pressure (D_p) into a blade row. Because of the simplicity of the de Haller number, it is frequently used in preliminary design work but detailed blading uses the diffusion factor.

I- Effect Of Velocity Profile Distortions (Work Done Factor)

For a given inlet mass flow through the blade row, the effect of the wall boundary layers is to increase the axial velocities across the mainstream portion of the annulus and to decrease the axial velocity near the end walls. Because the values of velocity over the major portion of the annulus are greater, the mass averaged energy addition at the design flow is lower than the design, resulting in reduction of work capacity. Hence, the actual stage temperature rise is lower than that envisaged by the design values.

Due to the non-uniformity of the axial velocity profile there is a reduction in the effective work done in a blade row, the ratio between the actual work done and that achievable with uniform inlet conditions is known as **work done factor.** The work done factor varies with the stage in a compressor.

J- Radial Equilibrium Solution

The free vortex hub to casing design has been selected for all three compressors. The reason for choosing the free vortex is that it keeps the axial velocity more or less uniform in the radial direction. However, it gives substantial blade twist across from root to tip especially for low hub-tip ratio.

K- Space To Chord Ratio

An appropriate choice of space/chord ratio for compressor blading at subsonic speed is extremely important. For example, some correlations allow for the effect of space/chord ratio on incidence and deviation angles. Space/chord ratios in the range 0.7 to 0.8 are used at the rotor tip where the Mach number is higher. However, evidence suggests that near the design point, the efficiency tends to be slightly lower if the space/chord ratio is on the low side (and low aspect ratio), but the pressure rise and operating range are greater.

L- Aspect Ratio

The choice of aspect ratio is related to the number of blades, stage length, effect on secondary losses and vibration. The use of low values leads to an increase in the secondary and tip clearance losses. High values of aspect ratio lead to vibration problem. The choice of a suitable value depends on the application. For fans and the L.P. compressor, the inlet values may be chosen between 4 and 4.5. However, new design trends indicate aspect ratio between 1.5 and 2.5 for most subsonic and transonic designs. This is because large chord blades are more effective in the end wall regions and which are crucial in determining both efficiency and the stall point.

M- Tip Clearance

The effect of tip clearance on overall compressor performance is not straightforward simple. Some experiments showed that optimum tip clearance lies between 1 and 1.5 per cent of chord for rotors, compared with between 3 and 5 per cent for cascades. However, the optimum clearance for rotors is generally smaller than which can be tolerated for mechanical reasons.

Other experiments have shown that not only does the pressure recovery rise and efficiency fall as the tip clearance is increased, but the surge line moves a considerable way downwards a lower pressure ratio. In a multistage axial-flow compressor, the precise effect of tip clearance becomes confused because the performance of each blade row affects all those downstream. For instance, increased tip clearance yields extra blockage and extra loss in the front stages and the matching of downstream stages is altered. In the case of stators it is relatively easy to provide a shroud seal at the hub.

Amongst all experimental work in the literature the following rule may be draw for tip clearance: every 1 per cent increase in tip clearance results in about 2 per cent deterioration in efficiency. A good range to take tip clearance is between 0.5 and 1.5 per cent of chord at tip.

Turbine Aerodynamic And Mechanical Parameters

C- Degree Of Reaction

Degree of reaction represents the fraction of the stage expansion which occurs in the rotor. Normally the turbines are designed for reaction of 50% at blade mid height. For the last stage, an exit swirl angle less than 20 degrees is necessary to minimise losses in the jet pipe and propelling nozzle. In this circumstance, it might be necessary to use a degree of reaction somewhat less than 50% if a high value of ψ is also required. Negative value of reaction must be avoided since this would produce expansion in the nozzle followed by recompression in the rotor increasing the losses.

D- Number Of Stages

The number of stages is determined primarily from considerations of performance at the design point. Other factors which also have to be considered are: engine cost and weight, engine geometry and the number of blade rows needing to be cooled. The calculation of the number of stages depends on the choice of work split between turbine stages. The initial choice of work split is based on experience. From the mechanical point of view, the work splits have to be selected in order to provide enough camber and stiffness in the blading.

E- Hub-Tip Ratio

The correct choice of hub-tip ratio in the first and last stage is important. Good practice is to select a hub-tip ratio for the high-pressure first stage not above 0.875, to maintain low tip clearance annulus losses. The hub-tip ratio of the last (low-pressure) stage should not be below 0.6, for stress consideration.

F- Space-Chord Ratio (s/c)

The optimum value of the space-chord ratio in the NGV and rotor rows can be found using Zweifel's criterion. Zweifel's formula at blade mid height is given by:

$$C_L=2(s/C_{ax})\ COS^2\alpha_2\ (\tan\alpha_1 + \tan\alpha_2)$$

Where station 1 and 2 are inlet and outlet of the blade row, s is the pitch and C_{ax} is axial chord, then:

$$C_{L\ opt}= 0.8/\ (\sin^2\alpha_2)$$

Once the space-chord is determined the following should be checked:

1- That the pitch (s) is not so small that the blades can not be attached safely to the turbine disc rim.

2- That there is sufficient chord (c) for blade cooling.

3- That the number of blades is not more than 100.

The number of the blades with common multiples should be avoided in order to minimise the probability of vibration. The usual practice is to adopt an even number for the nozzle blades and a prime number for the rotor blades.

G- Aspect Ratio (h/c)

Low values of aspect ratio are likely to increase secondary loss for the nozzle row and secondary and tip clearance loss for the rotor row. High values of aspect ratio will, conversely, increase the probability of vibration problems and reduce heat transfer. Vibration problems caused by high aspect ratio blading can be significantly reduced using tip shrouds. Typical limits for aspect ratio are 1< AR< 5.5, where AR for the NGV is often lower than that for rotor, because of cooling requirements.

H- Exit Swirl Angle